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STUDY REPORT

BLOCK II SURVEYOR STUDY [U]

MARCH 1964

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JPL CONTRACT 950056

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HUGHES

HUGHES AIRCRAFT COMPANY
SPACE SYSTEMS DIVISION

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SSD 4135R

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CONTENTS

1. INTRODUCTION

Hughes Aircraft Company has performed a Block II Surveyor Study in accordance with Change Order 32 to Jet Propulsion Laboratory Contract 950056. The study was initiated on 27 January 1964. This report is submitted in fulfillment of the requirement of Paragraph 2 of Change Order 32.

For Block II Surveyor launches beginning early in 1967, launch vehicle injected weight capabilities of 2200 to 3200 pounds may be available. The primary effort of this study has been directed toward a determination of the weight of payload that can be delivered to the moon by Surveyor as a function of the increased separated weight capability. The entire range of 2200 to 3200 pounds was investigated; however, emphasis was given to the range from 2200 to 2600 pounds. Both 66-hour and 90-hour trajectories were considered.

To further increase payload capability, the capabilities provided by alternate designs for the main retro rocket engine were studied. The most significant changes considered were the use of beryllium propellant and a titanium case instead of the present Surveyor retro design.

The study was performed independent of payload. However, four types of mission were investigated to define payload capability for four general classes of payload. The four mission types are as follows:

1) Landing and limited survival. This type of mission would correspond to a payload that is self-sufficient, including independent power and communications. The basic bus would provide communications to initiate payload operation on the lunar surface, but would not provide a continuing support to the payload.

2) Thirty-day lunar survival. This is similar to the mission performed by A-21A, but with the greater payload weight capability provided by the Block II Surveyor.

3) Ninety-day lunar survival. This is the same as mission 2, but with 90-day survival on the lunar surface.

4) Two-year lunar survival. Same as above, but with 2-year survival. It is shown that this mission is not practical with present Surveyor basic bus design concepts.

The basic bus designs for the four missions would be essentially the same except for changes in the power subsystem.

Improved landing accuracy may be desired for Block II Surveyors. The improved accuracy that can be expected from the use of turn-around ranging data in orbit determination in conjunction with a second mid-course correction has been determined.

To determine further the capability of the Block II Surveyor, the gross effects on design and performance of extension of the landing area to 60° E longitude were determined. A landing at 60° E longitude corresponds to an approach displaced approximately 75 degrees from the vertical, as compared to a maximum of 45 degrees allowed for in the A-21A design.

Finally, the Block II Surveyor Study considered the addition of lunar surface mobility to the basic bus by means of liftoff from the surface and lateral translation following the initial landing.

An underlying ground rule for this study has been that a minimum amount of change be made in the basic bus design that will have been proven during the A-21/A-21A program. This results in minimum development risk, maximum reliability through the use of proven hardware, and minimum cost. Unfortunately, some changes in the basic bus are required to utilize higher injected weight capability of the launch vehicle. In addition, a small number of additional basic bus changes that enhance payload capability or flexibility are discussed in this report.

During the course of the study, it has been necessary to create additional ground rules or make certain assumptions. The most significant of these are the following:

- A parking-orbit trajectory will be employed.
- No change will be made in the Surveyor/Centaur interconnect or in the Surveyor nose fairing.
- The lunar model specified for A-21 and A-21A is applicable to Block II Surveyor.
- There will be no degradation of the interface presented to the payload by the basic bus, as defined in Hughes specification 239503C, and supplemented by Hughes document 2256/70, a booklet of descriptive material presented to the JPL Lunar Roving Vehicle contractors at a briefing on 24 October 1963.

It is not the purpose of this study to define a spacecraft configuration. However, the expected capability described herein for a Block II Surveyor when combined with a statement of launch vehicle capability and definitions of payload and mission objectives will permit the configuration phase of a Block II Surveyor spacecraft to be undertaken.

2. SUMMARY

Payload weight capabilities have been determined for 18 different spacecraft/mission configurations. Before payload weight capabilities could be determined, it was necessary to determine spacecraft dry landed weights.

Various Surveyor Block II design alternatives are evaluated in Section 3 to determine propellant loading requirements and dry landed weights. The basic ground rules adopted were as follows:

1) Minimum or no change from Surveyor Block I unless absolutely necessary, e.g., the radar sensor constraints on the terminal descent were assumed unchanged.

2) Parking orbit injection by the Atlas/Centaur launch vehicle was assumed, with launches in the time period 1967 through 1969.

3) A midcourse correction capability of any magnitude up to 30 meters per second was provided.

4) The existing Surveyor/Centaur interface including interconnect structure and shroud was preserved.

The various design approaches included propulsion system and trajectory alternatives. There were:

1) Two types of vernier engine systems. The first considered had a restricted throttling range, 3.5:1, with resultant spacecraft thrust-to-weight ratios the same as for Surveyor Block I. The other vernier system was assigned an extended thrust range of 20 to 180 pounds per chamber, which is adequate for all designs considered.

2) Steel (present design) and titanium (improved design) main retro cases were considered.

3) The present main retro propellant with aluminum additive was evaluated with respect to an improved main retro propellant with beryllium additive.

4) Two trajectory classes were included in the study: the 66-hour class of transit trajectory currently employed for Surveyor Block I, and the 90-hour class.

A comprehensive study of the impact velocities for which the Surveyor Block II would need to be designed was undertaken. The results indicated that for parking orbit trajectories in the years 1967 through 1969, a range of impact velocities of 2611 to 2687 meters per second for 66-hour transits, and of 2512 to 2585 for 90-hour transits would occur. Spacecraft propellant requirements were determined, predicated on the maximum velocities of 2687 and 2585 meters per second, consistent with Surveyor Block I design practice. The resulting dry landed weights, evaluated parametrically for the injected weight range of 2200 to 2600 pounds, ranged from about 620 to about 840 pounds. The dominant effects on dry landed weight were as follows:

1) Injected weight – roughly 35 percent of any increase in injected weight is realized in dry landed weight, although this coefficient varies with the particular design.

2) Vernier engine type – 15 to 16 pounds dry landed weight advantage for extended throttle range type.

3) Main retro case – titanium case provides 9 to 10 pounds more dry landed weight.

4) Main retro propellant – beryllium propellant provides approximately 30-pound increase in dry landed weight.

5) Transit time – the 90-hour class gives a 30 to 32-pound dry landed weight advantage if spacecraft injected weight is held constant. If the comparison between trajectory types is made on the basis of constant launch vehicle capability, about a 45-pound dry landed weight increase is obtained for 90-hour, as compared to 66-hour transit time.

In Section 7, payload weight capabilities are determined. The principal design variations considered were in the propulsion system; two different versions of the main retro engine were combined with two different vernier engine systems to provide four different propulsion system options. Dry landed weights for these options are taken from Section 3. Within each of these four propulsion groupings, spacecraft design changes in the electrical power and telecommunications systems are examined to illustrate the effects of combining the most likely design provisions. The effect of a 90-hour trajectory is presented in two of the cases.

Maximum payload weight capabilities for the eighteen configurations studied range from 157 to 318 pounds. The payload weight limit for each of the configurations is established by the maximum volume of main retro propellant that can be accommodated without change of basic spaceframe geometry or Surveyor/Centaur interface. Maximum injected weights for the 18 configurations range between 2486 and 2639 pounds. The 157-pound payload weight capability is provided when the heaviest power system is combined with the lowest performance propulsion system using a 66-hour trajectory. The 318-pound payload weight capability is provided when lunar surface survival and wide-band telemetry are not required, and the highest performance propulsion system is combined with a 90-hour trajectory.

Parametric curves are also presented to indicate how payload weight varies with the injected weight for each of the spacecraft design configurations studied.

Power systems considered for Block II fall in two general categories: solar panel-battery, and solar panel-battery-RTG. In all cases, 100-percent battery redundancy is provided for reliability. Batteries are of the sealed secondary silver-zinc type presently used, but are sized differently for the landing/limited survival type of mission and for the 30/90-day survival missions with and without RTG. The RTG considered is the SNAP-11. For 30/90-day survival missions, the RTG system provides a weight saving of approximately 35 pounds, and also permits greater mission flexibility. Because of the use of redundant batteries, the probability of successful flight and landing for all missions considered is in excess of 80 percent. With the solar panel-battery-RTG system, probability of lunar surface survival is estimated at 0.86 for 30 days and 0.67 for 90 days. A 2-year survival mission is not considered practicable with present spacecraft design concepts.

In Section 9, it is shown that landing errors for one midcourse correction are almost entirely a result of errors in the execution of the corrections, and range from 18 kilometers (1σ) at zero degree incidence to 27 kilometers at 45-degree incidence. The landing error after a second midcourse correction depends in general on both orbit determination and execution errors. For 66-hour trajectories, using doppler and angular data (but not range data), 1σ errors will vary from 2.4 to 8.2 kilometers; with range data, the errors are reduced to 1.8 kilometers to 2.7 kilometers, and result primarily from execution errors. For 90-hour trajectories, the landing errors will be somewhat greater than for the 66-hour case if the second maneuver is performed during the second Goldstone pass, but slightly smaller if the maneuver is executed during the third Goldstone pass. To permit the acquisition of range data, modifications must be made in the spacecraft transponder. In addition, provisions must be incorporated to permit use of the spacecraft planar array antenna for receiving to provide ample signal-to-noise ratio of the range-code-modulated signal transmitted from earth. It is questionable whether the improvements of landing accuracy on the order of two or three provided by ranging justify these spacecraft modifications.

Two methods for extending the Surveyor landing area capability are examined in Section 10. Method A involves in-flight canting of the AMR antenna. Method B has, in place of canting the AMR, the requirement of a preretro attitude maneuver. Both methods also require the addition of a postburnout attitude erection maneuver. It was found that both methods are feasible but Method A has a considerable payload advantage (14.5 pounds for a 2150-pound separated weight) over Method B. The equipment additions and changes required for Method A, with the exception of the canting mechanism, are nearly identical with those of B. The reliability estimates favor Method B only very slightly. It is therefore concluded that in-flight canting of the AMR antenna is preferred.

In Section 11, it is concluded from a study of liftoff and translation of the Surveyor spacecraft along the surface of the moon that this does not appear to be a feasible means for accomplishing the site certification mission.

3. DETERMINATION OF DRY LANDED WEIGHT

A major task was the determination of Surveyor Block II dry landed weight under a wide range of separated weights, propulsion system alternates, and trajectories. Section 3 presents the ground rules and assumptions used, considerations regarding the various alternatives, and fuel loadings and dry landed weights for the various cases studied.

ALTERNATIVES

The following alternatives and injected weight variations were employed during the study.

1) Injected weight. The original study outline specified an injected weight range of 2200 to 3200 pounds. The lower end of the range corresponds approximately to present Atlas/Centaur capability, while the upper end suggests the use of a considerably higher performance booster. The original calculations were made for this entire injected weight range, but during the course of the Block II study, refined inputs from JPL made it clear that the upper portion of the injected weight range was unattainable because of the absence of booster capability and that attention should be directed to the injected weight range from 2200 to 2500 or 2600 pounds. As pointed out later, this limit corresponds approximately to the range of main retro loading limits which preserve the present interface with the Centaur vehicle, and which leave the Surveyor spaceframe essentially unchanged.

2) Vernier engine type. Two types of vernier engine systems were included in the study. One resembled the A-21/A-21A vernier engine system in that the throttling range was the same (maximum to minimum thrust ratio of 3.5:1), and the specific impulse was the same. However, the actual thrust range (30 to 104 pounds thrust per chamber) of this engine system was inappropriate to meet the adopted constraints at the higher injected

weights; therefore these thrust levels were used at an injected weight of 2200 pounds only, and were scaled to injected weight at the higher levels. Thus, the actual use of such an engine type for Surveyor Block II implies a totally new engine development, or at the very least, major redesign of an existing system. The engine system is referred to as the restricted throttle range vernier engine system (RTRVES).

The second vernier engine system selected for study was assigned the performance characteristics (thrust range and specific impulse) of the STL MIRA 180 system. Thus, the minimum thrust was assumed to be 20 pounds, and the maximum 180 pounds per chamber, a 9:1 throttling range. This range of thrusts is adequate for all the injected weights considered. This engine system is referred to as the extended throttling range vernier engine system (ETRVES). The problems involved in assuring compatibility between such an engine system and a Surveyor spacecraft are developed in detail in a Hughes report.*

3) Main retro case. As a result of studies made by JPL, it appears feasible to substitute titanium for some of the steel presently used in the Block I Surveyor main retro case, for a total case weight savings of 11 pounds. Accordingly, performance results were calculated for both types of case, assuming 11 pounds as the weight difference.

4) Main retro propellant. Since the inception of the Surveyor retro development program, considerable progress has been reported in the technology of solid engine propellants employing beryllium as an additive. These propellants deliver higher specific impulse than do conventional propellants with aluminum additive. Nevertheless, there are problems of some significance associated with the employment of beryllium propellants; these are discussed in another portion of this section. For study purposes, system performance was evaluated using both aluminum and beryllium propellants.

5) Transit time. The Surveyor Block I spacecraft is injected into earth-moon trajectories whose time of flight falls into the 66-hour transit time class, i. e., the spacecraft arrives at the moon at some time during the third Goldstone-spacecraft view period after injection. However, when the time of flight is longer, the required Centaur injection energy and the lunar impact velocity are both decreased. Therefore, trajectories where arrival takes place during the fourth Goldstone view period, i. e., those trajectories falling into the 90-hour transit time class, offer the prospect of higher dry landed weights than 66-hour trajectories. Accordingly, 66 and 90-hour trajectories were considered in this study.

*"Evaluation of Vernier Thrust Chamber Assembly, Hughes Aircraft Company SSD 4104R, 17 February 1964.

ASSUMPTIONS AND GROUND RULES

A basic guideline throughout the study was that of a minimum change from Surveyor Block I. Only absolutely necessary changes were permitted to the radars, both the AMR and the RADVS. For the AMR, this meant design of the terminal descent trajectory so as not to increase the required maximum marking range of 60 miles. This introduced the requirement for constant main retro action time, with effects on the main retro grain design and thrust profile. The RADVS constraints on main retro burnout velocity were as follows:

1) The 700 fps linear doppler limit constrains the maximum nominal burnout velocity to 560 fps, except where further limited by the altimeter, as below.

2) Consistent with the 45-degree unbraked impact angle capability, the doppler beam signal-to-noise limit has the effect of limiting the minimum nominal burnout velocity to 330 fps.

3) The radar altimeter limit at the descent trajectory intersection with the descent curve mechanized in the flight control system is shown in Figure 3-1.

The first Block II launch is assumed to be in February 1967, with a total of 18 launches at 2-month intervals. This implies a last launch month, for design and planning purposes, of December 1969. Therefore, the years 1967 through 1969, inclusive, are the ones of interest here. Because the flight program does not begin until 1967, Centaur parking orbit capability and trajectories are assumed throughout.

Consistent with Surveyor Block I design and operational planning, it is assumed that the spacecraft must arrive at the moon while within view of the Goldstone Deep Space Station (DSS). To allow for commanding of the terminal maneuvers and acquisition, the spacecraft is constrained to arrive no earlier than 2 hours after the beginning of the Goldstone view period. To allow for critical postlanding operations, the spacecraft must arrive at the moon no later than 3 hours before the end of the Goldstone view period. These constraints are consistent with Surveyor Block I.

The capability must exist for the spacecraft not to make a midcourse maneuver if one is not required to correct injection guidance errors (i.e., burnout velocity limits during the terminal phase must be satisfied even without a midcourse correction), and the maximum spacecraft midcourse maneuver capability will be 30 meters per second. As pointed out in Section 9, the maximum second midcourse maneuver required is 1.2 meters per second, which has negligible effect on spacecraft design. Hence, the effect of a second midcourse maneuver (which is less than 1/2 pound of payload) was ignored.

The vernier fuel capability was not constrained by the present tank sizes. It was determined that the tanks could be located so that they could easily be increased to a total vernier fuel capability of about 260 pounds, more than adequate for any of the spacecraft designs considered here. Accordingly, vernier fuel capacity was not a constraint on spacecraft design. It was also assumed, as in Block I, that 4.2 pounds of the vernier propellant loaded would be unusable.

It was assumed that the restricted throttling range vernier engine system employs electromechanical actuated throttling valves, whereas the extended throttling range vernier engine system was assumed to employ vernier fuel as hydraulic fluid in hydraulic servoactuated throttling valves. Five pounds of fuel was assumed to be used and dumped for this purpose during a mission.

Up to 45-degree off-normal unbraked impact angle capability was provided. All propulsion sizing calculations were for vertical descent, the condition of maximum fuel consumption.

The various weights affecting dry landed weight were assumed to be as follows:

1) AMR, 8.9 pounds

2) Helium, 2.5 pounds, retained in spacecraft

3) Nitrogen, 4.5 pounds of which 2.5 pounds were assumed expended before main retro ignition

4) Main retro case weights as per Figure 3-2.

Throttling ranges used for the vernier engines were discussed on pages 3-1 and 3-2.

Restricted Throttling Range Vernier Engine System

$$\text{Maximum thrust} = 104 \text{ pounds per chamber} \times \frac{\text{Injected weight}}{2200 \text{ pounds}}$$

$$\text{Minimum thrust} = \frac{\text{Maximum thrust}}{3.5}$$

Extended Throttling Range Vernier Engine System

Maximum thrust = 180 pounds per chamber

Minimum thrust = 20 pounds per chamber

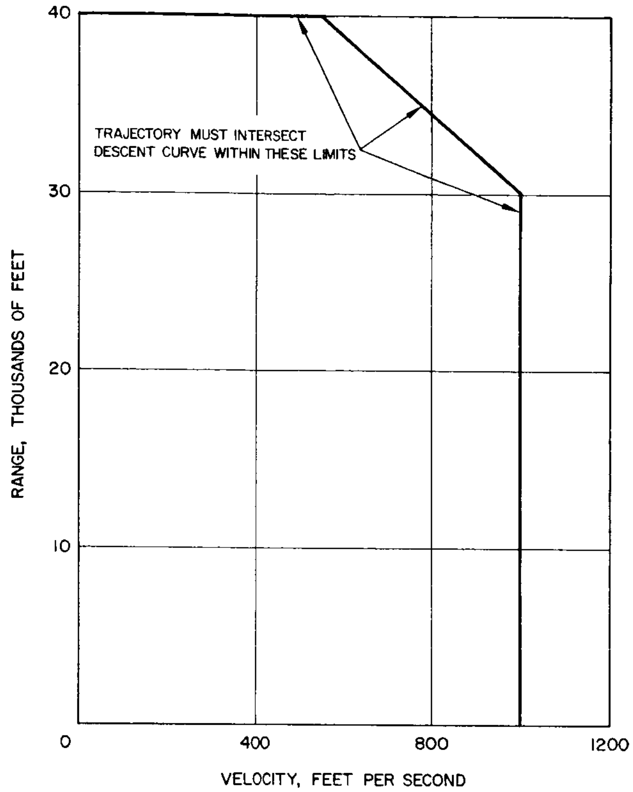

Figure 3-1. Radar Altimeter Constraints

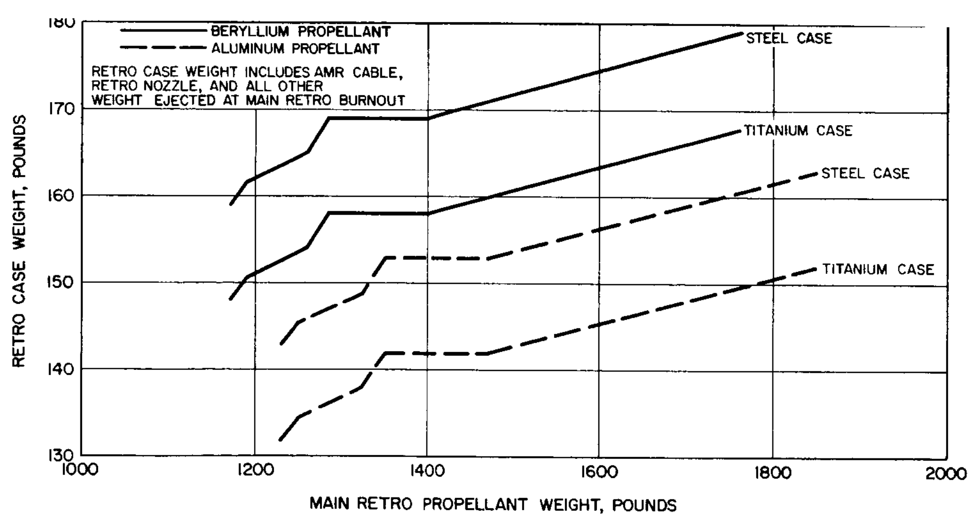

Figure 3-2. Main Retro Case Weights

Vernier engine specific impulse values were assumed to be those shown in Figures 3-3 and 3-4. Retro engine specific impulse values were assumed to be 290 seconds for aluminum and 307 seconds for beryllium propellants.

Vernier engine thrusts during the main engine burning sequence were assumed to be those shown in Table 3-1.

TABLE 3-1. ENGINE THRUSTS DURING MAIN ENGINE BURNING SEQUENCE

Vernier System	Vernier Engine Thrust, pounds	
	During Main Retro Burning	During Main Retro Case Separation
Restricted throttling range	$200 \times \frac{\text{Injected weight}}{\text{2200 pounds}}$	$280 \times \frac{\text{Injected weight}}{\text{2200 pounds}}$
Extended throttling range	330	504

THE VARIATION OF LUNAR IMPACT SPEEDS, 1967 TO 1969

Introduction

An extensive investigation was undertaken to determine the magnitude and variation of lunar impact speeds for 66- and 90-hour flight times during the time period 1967 to 1969. This information was required to design the main retro and vernier propulsion systems for the Surveyor Block II mission. The data from a near-earth conic trajectory program was correlated with that obtained from an N-body integrating program. Once the correlation was obtained, a large number of cases were run on the conic program to determine the variation in impact speed. A knowledge of the angular relationship between the line of nodes and the line of apsides of the lunar orbit was found to be useful, and was used in determining when maximum and minimum impact speeds occur.

Ground Rules

The following ground rules were used in the impact velocity study:

1) The Atlas/Centaur boost vehicle as previously defined for the parking orbit type mission is used.

2) Parking orbit ascent trajectories are used with the minimum coast time of zero and maximum coast time on the order of 35 minutes, the maximum required for the type of parking orbit trajectories considered.

3) Launches take place from Cape Kennedy with launch azimuth restricted to lie between 90 and 114° E of the true north.

4) A total of 18 launches are being considered, beginning in February 1967, with an every other month schedule. The time period of interest is therefore February 1967 through December 1969.

5) Flight times on the order of 66 and 90 hours are considered.

6) Lunar arrival must take place between 2 hours after Goldstone rise and 3 hours before Goldstone set. This is identical to the arrival time constraint used for Block I Surveyor trajectory design. Goldstone rise and set refer to the times of the moon's rise and set with respect to the Goldstone DSS as limited by a 5-degree elevation angle, land mask, or antenna position restrictions.

7) No lighting constraints are considered for the determination of velocity limits. Arrival may take place at any time during the lunar month.

Analysis

The variation in impact velocity is influenced significantly by the allowable variation in arrival (and flight time) as dictated by the requirement of Goldstone visibility of the arrival at the moon. The duration of the Goldstone lunar view period is shown in Figure 3-5 as a function of lunar declination. The allowable variation in arrival time is the duration of the Goldstone view period less 5 hours for the 2-hour preimpact and 3-hour postimpact constraints.

The variation in flight time (from injection to impact) as shown in Figure 3-6 is less than the variation in arrival time. This is because the earliest arrival must be launched at an azimuth of 90 degrees (the earliest launch) and the latest arrival must be launched at an azimuth of 114 degrees (the latest launch). The difference between the increments in arrival time and flight time consists of the increment in injection time between the two azimuths.

It should be axiomatic that the allowable variation in flight time will be used to reduce the variation in impact speed. In other words, on days when the impact speed is low (at, say a fixed point in the view period) the

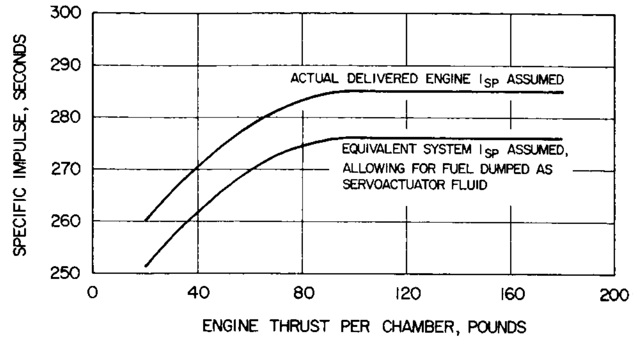

Figure 3-3. Extended Throttle Range Vernier Engine System (ETRVES) Specific Impulse versus Thrust

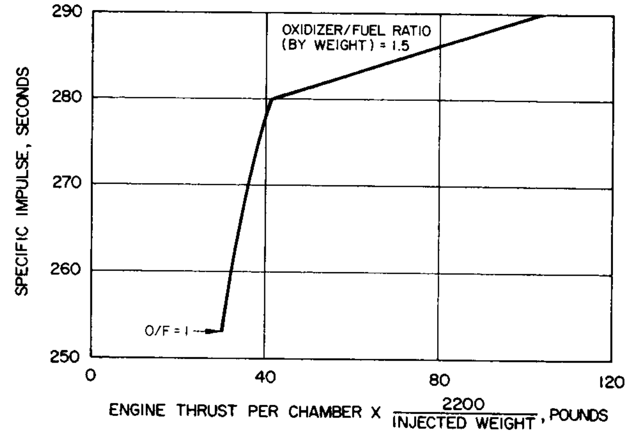

Figure 3-4. Restricted Throttle Range Vernier Engine System (RTRVES) Specific Impulse versus Thrust

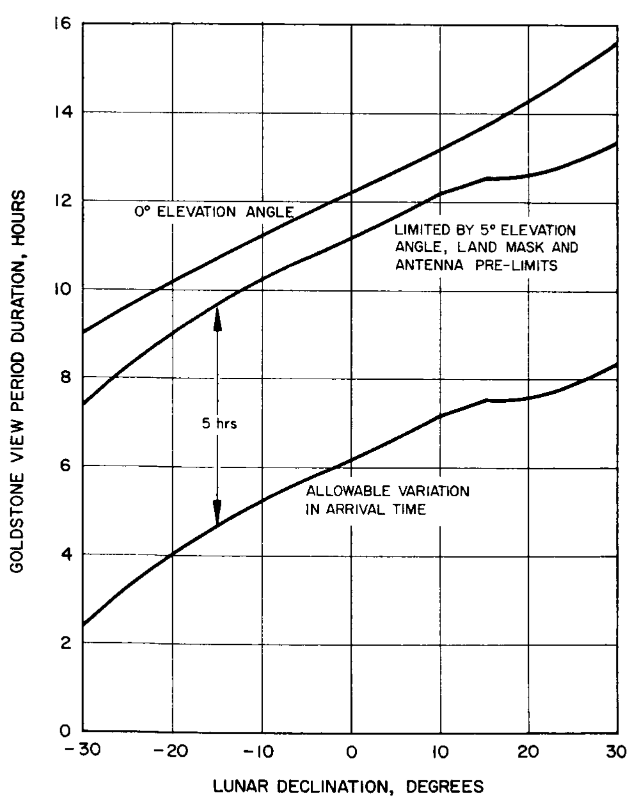

Figure 3-5. Variation of Goldstone View Period Duration with Lunar Declination

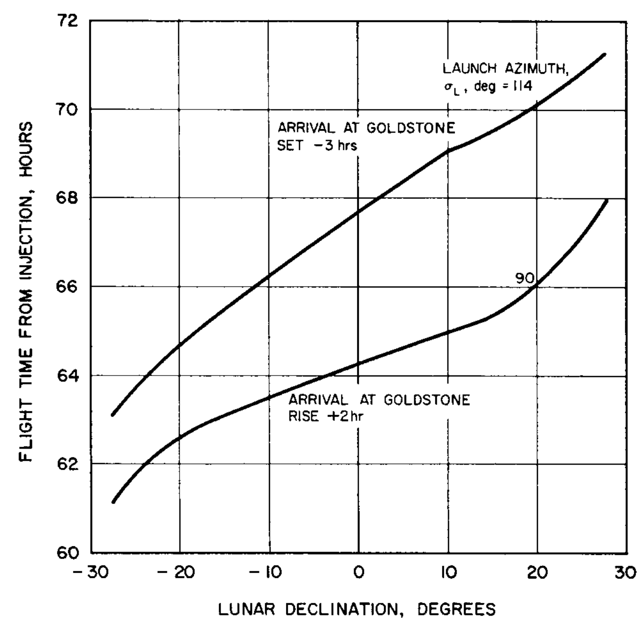

Figure 3-6. Variation of Flight Time with Lunar Declination

arrival will be biased towards Goldstone rise, thereby increasing the injection energy and impact speed; on days when the impact velocity is high the arrival will be biased towards Goldstone set, thereby reducing the injection energy and impact speed. Maximum impact speeds therefore occur at the 3-hour postimpact constraint point in the view period.

Two computer trajectory programs were used in the velocity study. The first is a near-earth conic program which assumes the trajectory to be a conic section from injection to the intersection of the trajectory plane with the lunar orbit plane. At this point the vector operation *

$$\overrightarrow{V_{PM}} = \overrightarrow{V_{\infty}} = \overrightarrow{V_{PE}} - \overrightarrow{V_{ME}}$$

is performed to determine the spacecraft velocity with respect to the moon (lunar approach velocity, with respect to a massless moon, V_{∞}). The second computer program is an N-body integrating program which considers all significant perturbations on the trajectory.

An investigation was made of the differences between the lunar approach velocities calculated by the two programs. The results, shown in Table 3-2, indicate that the conic program can be used to predict the variation of approach speed. Further, the most significant difference between the program is a relatively consistent bias in speed. For 66-hour trajectories the conic approach speeds ranged from 21 to 28 meters per second higher than the N-body results, the average being 25 meters per second. For 90-hour trajectories the conic approach speeds ranged from 13 to 34 meters per second higher than the N-body results, the average again being 25 meters per second.

These results indicate that if the average difference is used as a correction, the conic program can predict approach speeds to about ±4 meters per second for 66-hour trajectories and ±10 meters per second for 90-hour trajectories. The corresponding accuracies in impact velocity should be about ±2 meters per second and ±4 meters per second (Figure 3-7).

Another tool which proved useful in this investigation is a knowledge of the lunar orbit, particularly the orientation of the line of apsides and the line of nodes (the line of nodes being defined as the intersection of the lunar

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*Where

$\overrightarrow{V_{PM}}$ = velocity of spacecraft with respect to moon

$\overrightarrow{V_{PE}}$ = velocity of spacecraft with respect to earth

$\overrightarrow{V_{ME}}$ = velocity of moon with respect to earth

TABLE 3-2. COMPARISON OF CONIC AND N-BODY TRAJECTORY RESULTS

Case	Impact Date	Time of Flight, hours	Launch Azimuth, degrees	Injection Energy per Unit Mass (C_3) – km^2/sec^2			Lunar Approach Velocity (V_∞), meters per second		
				Conic	N-Body	Δ	Conic	N-Body	Δ
1	2-21-67	66	90	-1.580	-1.570	+0.010	1156	1128	-28
2	6-1-67	66	114	-1.166	-1.110	+0.056	1172	1146	-26
3	6-10-67	66	114	-1.551	-1.561	-0.010	1123	1097	-26
4	6-17-67	66	114	-1.671	-1.604	+0.067	1211	1189	-22
5	6-26-67	66	114	-1.228	-1.181	+0.047	1158	1133	-25
6	6-29-67	66	114	-1.198	-1.147	+0.051	1161	1134	-27
7	12-25-67	66	114	-1.644	-1.574	+0.070	1225	1205	-20
8	2-12-67	90	90	-1.768	-1.709	0.059	891	867	-24
9	2-26-67	90	114	-2.158	-2.093	0.065	1028	1011	-17
10	12-7-67	90	90	-1.856	-1.798	0.058	867	835	-32
11	12-25-67	90	114	-2.085	-2.009	0.076	1014	992	-22
12	12-14-68	90	114	-2.044	-1.968	0.076	1017	996	-21
13	12-15-68	90	114	-2.085	-2.017	0.068	1015	992	-23
14	12-26-68	90	90	-1.879	-1.822	0.057	859	826	-33
15	8-5-69	90	90	-1.869	-1.838	0.031	852	818	-34
16	8-17-69	90	114	-1.921	-1.848	0.073	1001	988	-13
17	12-4-69	90	114	-1.955	-1.879	0.076	1008	987	-21
18	12-23-69	90	90	-1.841	-1.837	0.004	861	828	-33

orbit plane and the earth's equatorial plane). The line of apsides advances in an oscillating fashion at an average rate of 40 degrees per year with respect to the line of nodes, completing one revolution in 8.85 years. This means that apogee and perigee occur at changing declinations from month to month. This behavior is shown in Figure 3-8a for the time period 1967 to 1969. A more helpful way of viewing this phenomenon from a trajectory viewpoint is shown in Figures 3-8b and 3-8c. The variation of the lunar distance at the ascending and descending nodes as well as the apogee and perigee distances is shown in Figure 3-8b. Figure 3-8c shows the variation of the distance at maximum and minimum lunar declination. Early in 1967 apogee and perigee correspond to the ascending and descending node while early in 1969 they correspond to maximum and minimum declination. This information will be shown to have a bearing on the variation of the approach speed as well as the location of the maximum and minimum values.

Results - 66-Hour Flight Time

The conic program was used to calculate lunar arrival speeds for every day from February through December 1967 with arrival at the post-impact constraint (late arrival). Arrival speeds for the pre-impact constraint (early arrival) were calculated for February and December only. These data are presented in Figure 3-9. The dashed lines in these figures enclose those areas at low lunar declination where the parking orbit coast time is less than zero. Flights cannot occur in these areas; the dashed lines are shown to make the curves continuous. Figure 3-9 shows the phases of the moon and the lunar orbit situation. Some computed results from the N-body program have been plotted for a comparison with the conic results. These conic results have not been corrected by the increment of 25 meters per second in the approach speed.

In the beginning of 1967 the monthly maximum in approach speed occurs when the moon is between minimum declination and apogee or at the same time the injection energy (C_3) reaches a maximum. There is a second monthly peak, however, at the lunar descending node which, beginning in April, produces a larger speed than the first peak. This corresponds to the time that the descending node and perigee coincide as shown in Figure 3-8b. For the rest of 1967 and through 1969 the monthly maximum in speed will always occur at the descending node. For 1967 the maximum values reach peaks in June and December, the one in December being about 15 meters per second higher. Figure 3-8b shows that the lunar distance at the ascending node also attains peaks in June and December, the greater being in December. Therefore, the conclusion can be drawn that the greater the lunar distance, the greater the speed. This makes it relatively simple, using Figure 3-8b, to evaluate the maximum approach speeds for 1968 and 1969. The lunar distance at descending node reaches annual peaks in December 1968 and December 1969. The approach speeds are determined to be 1231 and 1257 meters per second in these months.

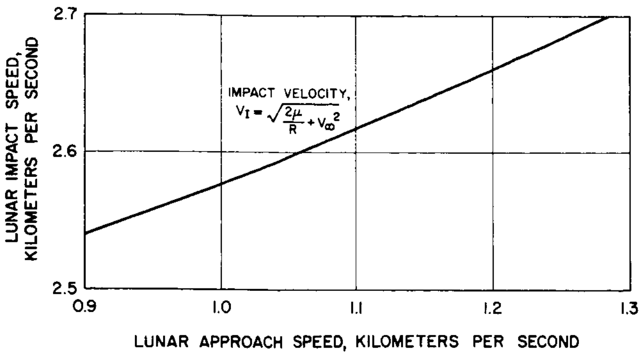


Figure 3-7. Variation of Lunar Impact Speed with Approach Speed

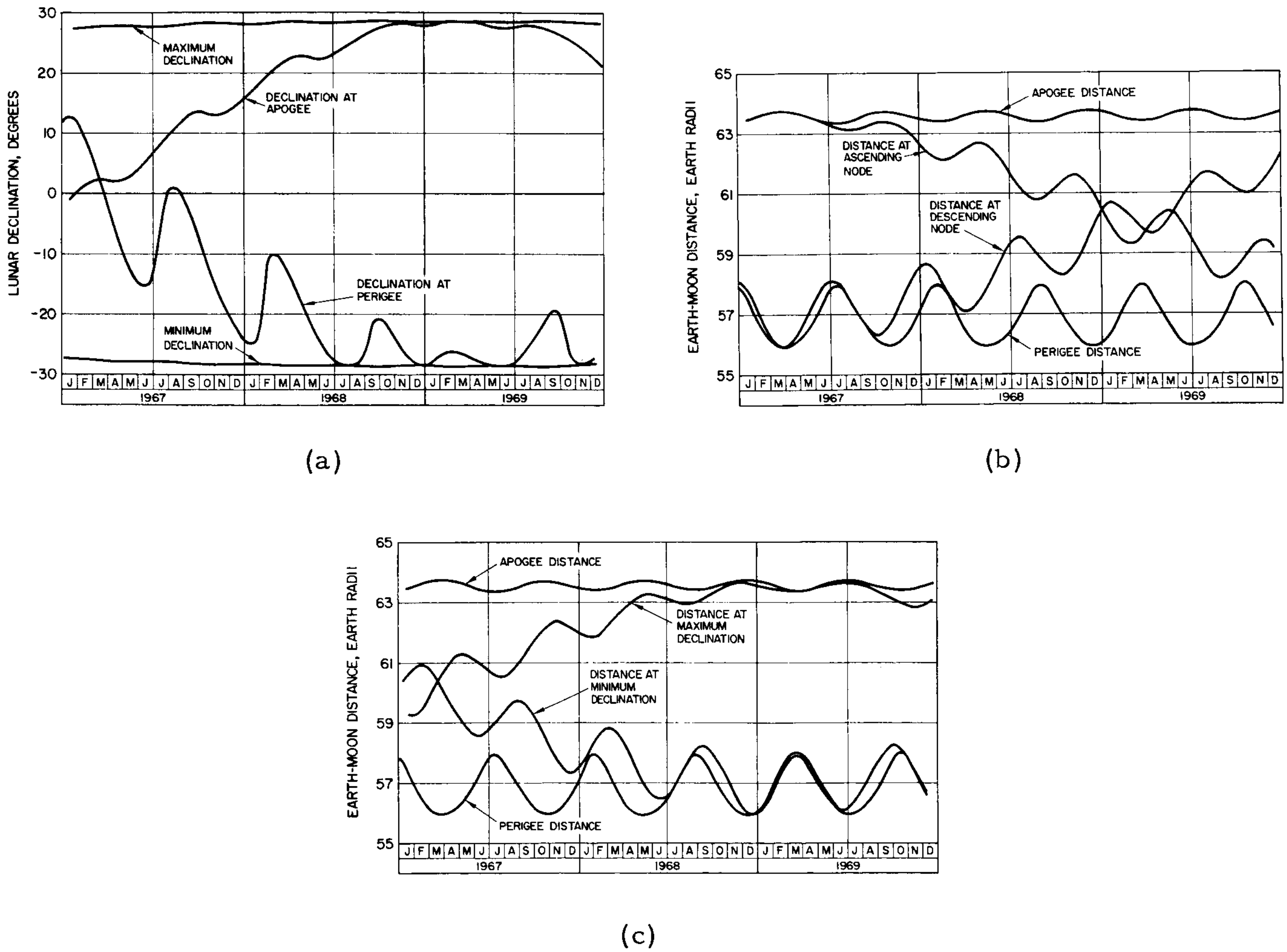
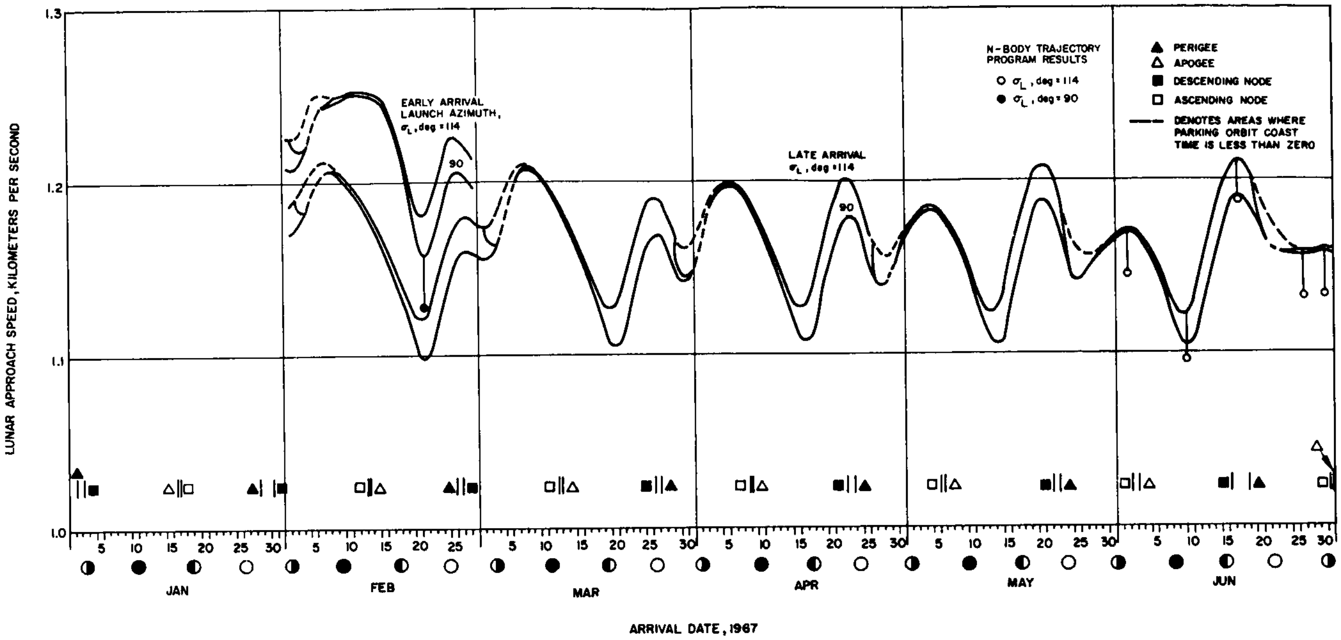
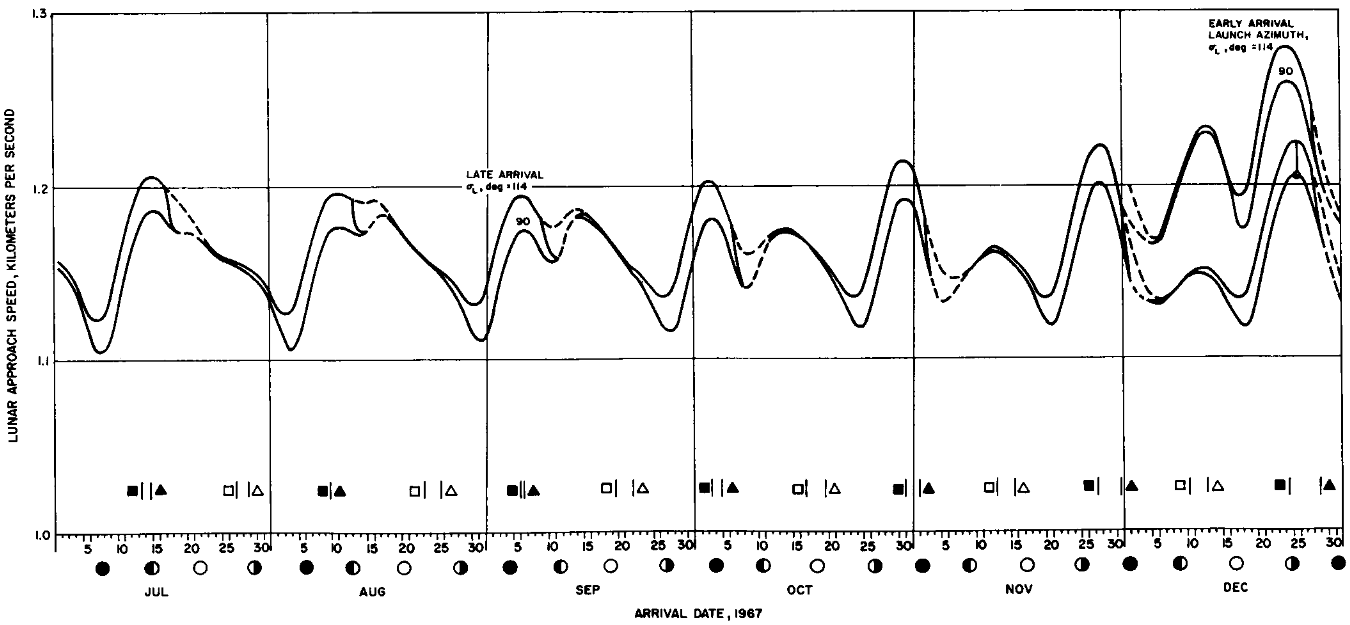


Figure 3-8. Lunar Orbit Characteristics



a) January-June 1967



b) July-December 1967

Figure 3-9. Variation of Lunar Approach Speed

The minimum values of approach speed during 1967 occur when the moon is at maximum declination. The lowest speed for the year occurs in February when the lunar distance at maximum declination is a minimum as shown in Figure 3-8c. By the end of 1968, however, maximum declination coincides with apogee and the monthly minimum in approach speed no longer occurs at maximum declination. At this time the minimum speed occurs at the ascending node. The minimum lunar distance at ascending node occurs in December 1968 and the corresponding approach speed is 1105 meters per second. A question arises as to whether earlier in the year a minimum speed occurs at maximum declination which is less than 1105 meters per second. Figure 3-8c shows that at no time during 1968 is the lunar distance at maximum declination less than in February 1967. Therefore, the speeds at minimum declination during 1968 must all be greater than 1128 meters per second,and 1105 meters per second must be the minimum for the year. The minimum speed for 1969 of 1085 meters per second occurs in August when the distance at ascending node is a minimum.

Annual maximum and minimum arrival and impact speeds for 66-hour trajectories during 1967 to 1969 are summarized in Table 3-3.

Results — 90-Hour Flight Time

The minimum and maximum values of approach speed for 90-hour trajectories are found to occur at the ascending and descending node respectively for the time period of interest. An important difference between the 66- and 90-hour flight times is that for the 90-hour case the maximum speed increases with decreasing lunar distance. In other words the annual maximum occurs when the lunar distance at the descending node is a minimum. With this information,Figure 3-8bis used to determine that the annual maximums in approach speed occur in March 1967, April 1968, and April 1969. The values of approach speed are 1020, 1019, and 997 meters per second respectively.

The minimum speed is found to decrease with decreasing lunar distance at the ascending node as it did for 66-hour flight times. The annual minimums occur in December 1967, December 1968, and August 1969. The values of approach speed are 835, 826, and 818 meters per second.

The annual maximum and minimum values of approach speed for 90-hour trajectories are summarized in Table 3-3.

MAIN RETRO ENGINE CONSIDERATIONS

A brief study was made of the possibilities of using the basic Surveyor main retro engine, loaded with more propellant, for the Block II system. The propellant weight range investigation was chosen to cover a vehicle weight range of 2200 to 3200 pounds; the aluminized propellant

TABLE 3-3. ANNUAL MAXIMUM AND MINIMUM LUNAR IMPACT SPEEDS 1967 TO 1969

Year	Time of Flight, hours	Impact Date	Launch Azimuth, degrees	Arrival	Lunar Approach Velocity (V_∞), meters per second	Lunar Impact Speed (V_I), meters per second
1967	66	2-21-67	90	Early	1128	2629
	66	12-25-67	114	Late	1205	2663
1968	66	12-14-68	114	Late	1231	2675
	66	12-23-68	90	Early	1105	2620
1969	66	8-1-69	90	Early	1085	2611
	66	12-31-69	114	Late	1257	2687
1967	90	3-26-67	114	Late	1020	2585
	90	12-7-67	90	Early	835	2518
1968	90	4-12-68	114	Late	1019	2584
	90	12-26-68	90	Early	826	2515
1969	90	4-2-69	114	Late	997	2576
	90	8-5-69	90	Early	818	2512

weights involved were 1250, 1324, 1420, 1440, 1446, 1470 and 1850 pounds. Beryllium-fuelled engines loaded to the same volume limits were also included in the study. The pertinent design criteria included a maximum thrust requirement, derived for each loading by direct scale-up of the current 2150 pound spacecraft requirement, maintenance of the current A-21 retro action time, and minimum envelope perturbation.

The maximum thrust criterion is based on the ability of the Surveyor attitude control system to compensate for main retro thrust misalignment moments using differential vernier throttling, while the retro action time limit is imposed to maintain compatibility with the current 60-mile maximum AMR marking range. Finally, since the configuration constraint adopted is that no changes will be considered in the Centaur/Surveyor interconnect structure, the Centaur shroud, or the Surveyor spaceframe basic geometry, it is necessary for the combined main retro/AMR assembly to exhibit no length increase or case diameter change. Referring to Table 3-4, the maximum configuration allowed under conditions of no AMR antenna change is Configuration 5, allowing a maximum of 1446 pounds of aluminum propellant or 1377 pounds beryllium propellant. With a planar array antenna for the AMR, a 5.4 inch engine length increase is allowable, resulting in Configuration 6. This allows maxima of 1470 pounds of aluminum propellant and 1400 pounds of beryllium propellant. Section 7 contains a more detailed discussion regarding the problem of main retro installation while maintaining the present Centaur/Surveyor interface, and discussion of possible AMR antenna changes which allow increase of retro nozzle length.

The significant characteristics of each configuration are listed in Table 3-4. The first and second involve up-loaded A-21 designs, while the remainder involve the A-25 design. The overall decrease in length noted in designs 3 and 4 resulted from decreases in throat area which increased the expansion ratios beyond the current 53/1 (which corresponds to a delivered specific impulse of 290 seconds). As this increase is not required, the nozzle length was reduced accordingly. In design 5 the nozzle was located 1 inch aft of the current position, exactly compensating for the length change incurred in design 4; propellant was added, bringing the propellant weight up to 1446 pounds.

Designs 6 through 9 indicate the envelope tradeoff involved in selecting the cutback technique to be employed in providing space for the submerged section of the nozzle. Not indicated in the table, however, is the fact that a conical cutback will induce a drastic departure from the current neutral thrust time trace characteristic of the A-25 grain. The trace characteristic of the conical cutback will have a definite peak at the midpoint and a considerably longer tailoff. This is not acceptable with respect to spacecraft system performance, and the conical cutback was not further considered.

Design 8 and 9 differ from all others listed in that a 7.6-inch cylindrical section has been inserted at the girth of the case. The inert weight

TABLE 3-4. MAIN RETRO ENGINE PARAMETERS, SURVEYOR BLOCK II*

Design	Al Propellant Weight	Al Propellant Change in Inert Weight,** pounds	Change in Length (Overall),*** inches	Maximum Chamber Pressure	Maximum Thrust	Action Time	Al Propellant Delivered I_{sp}	Corresponding Be Propellant Weight	Be Propellant Delivered I_{sp}	Comment
1	1250	+2.5	0.0	600	9,200	42.01	290	1190	308	A-21 configuration
2	1324	+6.0	0.0	740	10,800	42.03	290	1260	308	A-21 configuration (maximum loading)
3	1420	+10.0	-1.6	820	11,400	41.25	290	1354	308	Shorten nozzle
4	1440	+10.0	-1.0	820	11,600	42.01	290	1372	308	Shorten nozzle
5	1446	+10.0	0.0	820	11,600	42.01	290	1377	308	Pull nozzle 1.0 inch
6	1470	+10.0	5.4	820	12,040	42.03	290	1400	308	Step cutback
7	1470	+10.0	3.4	820	12,100	42.01	290	1400	308	Conical cutback – not satisfactory for Surveyor use
8	1850	+20.0	+14.6	820	15,440	42.03	290	1762	308	Step cutback; increase throat area, insert 7.6 inch cylinder section
9	1850	+20.0	+12.6	820	15,500	42.02	290	1762	308	Conical cutback; increase throat area, insert 7.6 inch cylinder section–not satisfactory for Surveyor use.

*Be configurations are 10 pounds heavier in case and nozzle, plus 6 pounds for thermal control. (Figures 3-2 and 3-3.)

**Assumed current (A-21) inert weight = 142.9.

***Change from present Surveyor Block I envelope restrictions.

increase is therefore proportionally greater. The cylindrical section did not need to be employed on any of the configurations considered.

The inert weight increase for all designs loaded with beryllium propellant is that listed for the corresponding aluminized propellant design, plus 10 pounds. The additional weight is necessitated by the higher flame temperature (requiring additional insulation) and different erosion characteristics (requiring a tungsten alloy throat insert) of the propellant formulation. To hold thrust level variations within the present system specifications, thermal control provisions amounting to 6 pounds of insulation and heaters must be added when beryllium propellant is used, since beryllium propellant exhibits approximately twice as much variation in thrust with temperature as does the aluminum propellant; this results in a 16-pound total increase.

It should be recognized that the indicated increases in chamber pressure are required to maintain the A-21 action time. The current case, nozzle closure, and nozzle designs are perturbed. All changes of this type necessarily imply a limited development/qualification program. The incorporation of a beryllium propellant implies basic development, including the design of an appropriate nozzle contour and throat insert.

The basic assumptions underlying Table 3-4 are as follows:

Aluminum propellant density	0.0634 lb/in^3
Beryllium propellant density	0.0604 lb/in^3
A-21 retro inert weight	142.9 pounds (includes AMR cabling)
A-21 retro delivered specific impulse	290 seconds
Beryllium propellant specific impulse	308 seconds (307 seconds used in dry landed weight calculations)
Pressure vessel safety factor	1.15
Steel case material ultimate	260,000 psi/min (Ladish D6AC)

COMPUTATION OF PROPELLANT LOADINGS AND DRY LANDED WEIGHTS

A newly developed design program was used for the computation of the required retro and vernier engine propellants. This program performs automatically a sequence of operations formerly requiring a sequence of computer runs over a period of several days.

The procedure for the main retro sizing remains unchanged. The retro is sized for maximum burnout velocity with zero midcourse correction and maximum approach velocity (2687 meters per second for 66-hour transits, 2585 meters per second for 90-hour transits). The maximum deceleration during the vernier phase is then computed, based on the maximum burnout weight and the vernier engine thrust capability. A check is made to assure that altimeter limits are not violated. If they are, the procedure is repeated for a lower burnout velocity.

The vernier fuel requirement is calculated in the same computer run by means of a Monte-Carlo simulation of the vernier descent in which 1000 separate fuel computations are made, with all random quantities being individually generated. The optimum ratio of oxidizer to fuel is determined such that on a conservative basis, the least total propellant yields a 99 percent probability of not running out of oxidizer or fuel. The results presented in Figures 3-10 and 3-11 were obtained. Dry landed weights are calculated by subtracting the weight of all expendables (including helium) from injected weight.

DISCUSSION OF RESULTS

The results indicated in Figures 3-10 and 3-11 show considerable variations in dry landed weight under the various conditions assumed. First, the propellant loadings, and consequently the dry landed weight, are almost linear with injected weight. This result has been consistently obtained for all Surveyor parametric studies. In addition, the various alternatives considered lead to relatively consistent differences in dry landed weight, and these effects may be superimposed. In comparing 90-hour transit times to 66-hour transit times, the comparison may be made for equivalent booster capability as well as at a constant injected weight. For the parking orbit trajectories assumed, there is a difference of approximately 30 meters per second between the required injection velocities for the two transit times, the 90-hour trajectory requiring the lower energy. For the Centaur launch vehicle, this corresponds to an injected weight difference of approximately 40 to 45 pounds. Hence, on the order of 14 pounds of spacecraft dry landed weight increase will be available because of the decrease in injection energy requirements alone, holding booster capability constant. In addition, there is an even larger increase in dry landed weight at constant injected weight due to the impact velocity decrease with 90-hour compared to 66-hour trajectories. Table 3-5 lists the effect of the various alternatives on dry landed weight.

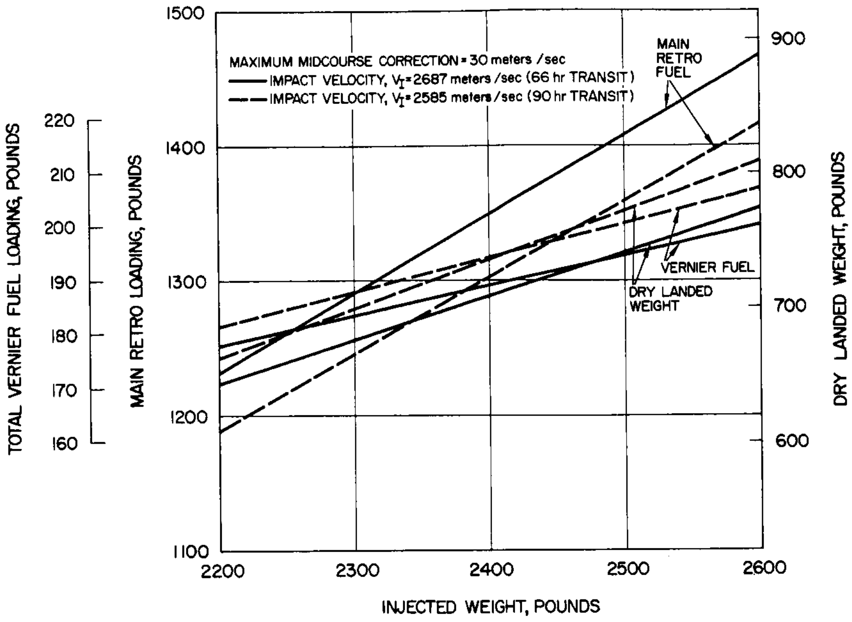
TABLE 3-5. EFFECT OF PROPULSION AND TRAJECTORY ALTERNATIVES ON DRY LANDED WEIGHT

Constant Spacecraft Injected Weight

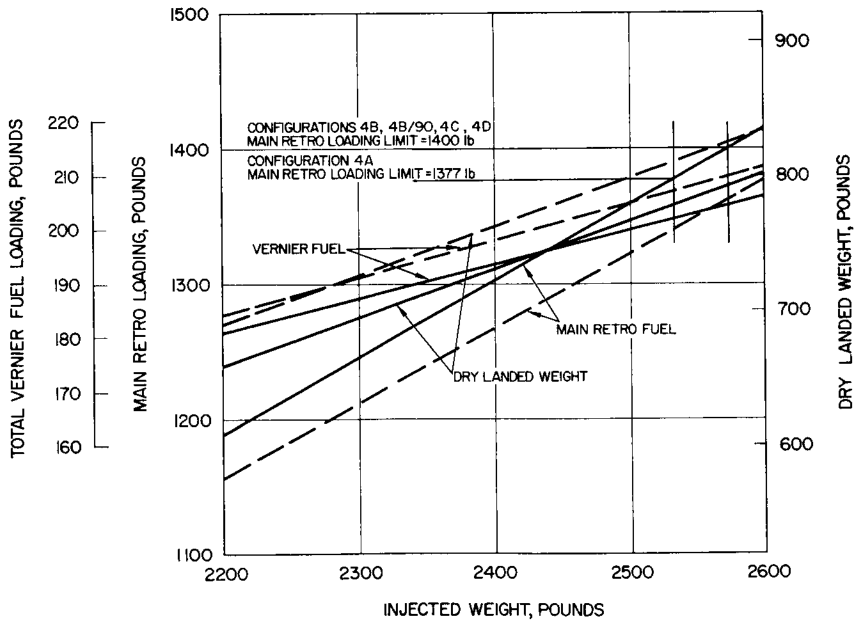
Alternative	Effect on Dry Landed Weight
Extended Throttle Range Vernier Engine System versus Restricted Throttle Range Vernier Engine System	15-to 16-pounds dry landed weight increase for ETRVES
Titanium versus steel main retro case	9-to 10-pounds dry landed weight increase with titanium case
Beryllium versus aluminum main retro propellant	Approximately 29-pounds dry landed weight increase due to beryllium propellant
90-hour versus 66-hour transit time	30-to 32-pounds dry landed weight increase for 90-hour trajectories at constant injected weight Approximately 45-pounds total dry landed weight increase for 90-hour trajectories with constant Centaur capability

The linearity of the weight relationships is illustrated by taking the average sum of the four dry landed weight increases at constant injected weight in Table 3-5. This total is 85 pounds. By comparison, the dry landed weight for a 2600-pound injected weight spacecraft using an ETRVES, titanium retro case, beryllium fuel, injected into a 90-hour trajectory is 835.5 pounds; for a 2600-pound spacecraft using a RTRVES, steel retro case, aluminum propellant, and a 66-hour trajectory it is 749.7 pounds. The dry landed weight difference is 85.8 pounds.

A comparison of the results of the present Block II study with the A-21A Surveyor spacecraft is also significant. Table 3-6 summarizes the pertinent parameters regarding the two systems. The dry landed weight differences are readily reconcilable below:

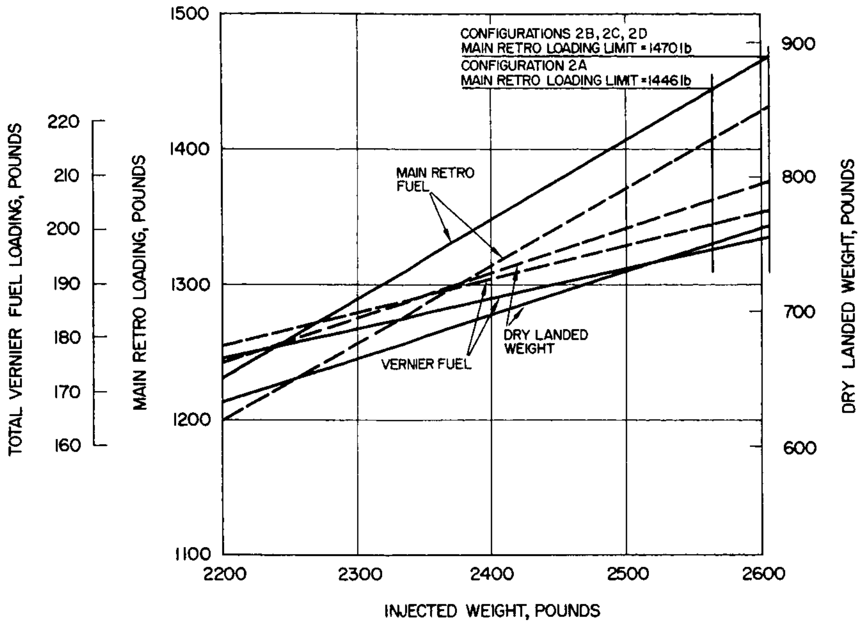


a) Titanium Main Retro Case
Aluminum Main Retro
Propellant

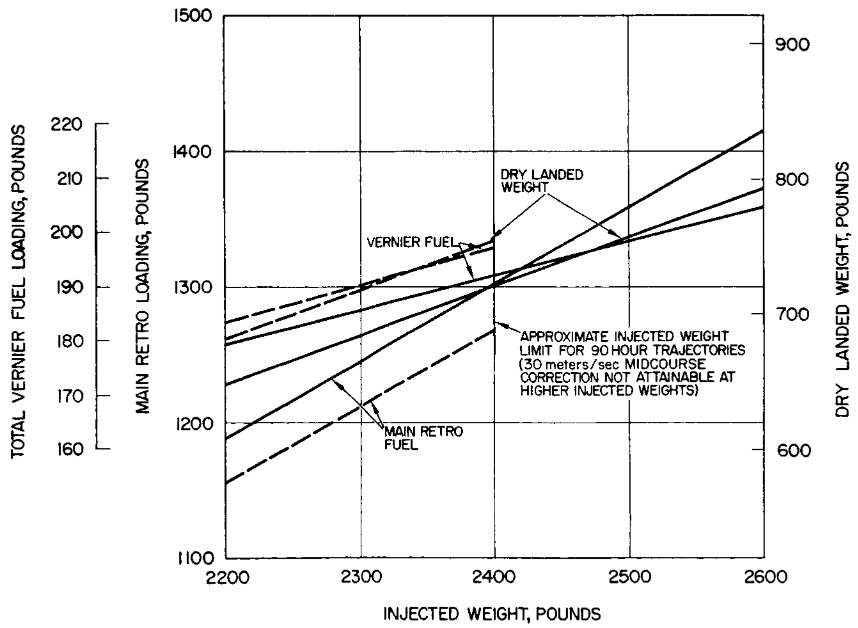


b) Titanium Main Retro Case
Beryllium Main Retro
Propellant

Figure 3-10. Dry Landed Weights and Required Propellants
Extended throttle range type vernier engines (9 to 1)

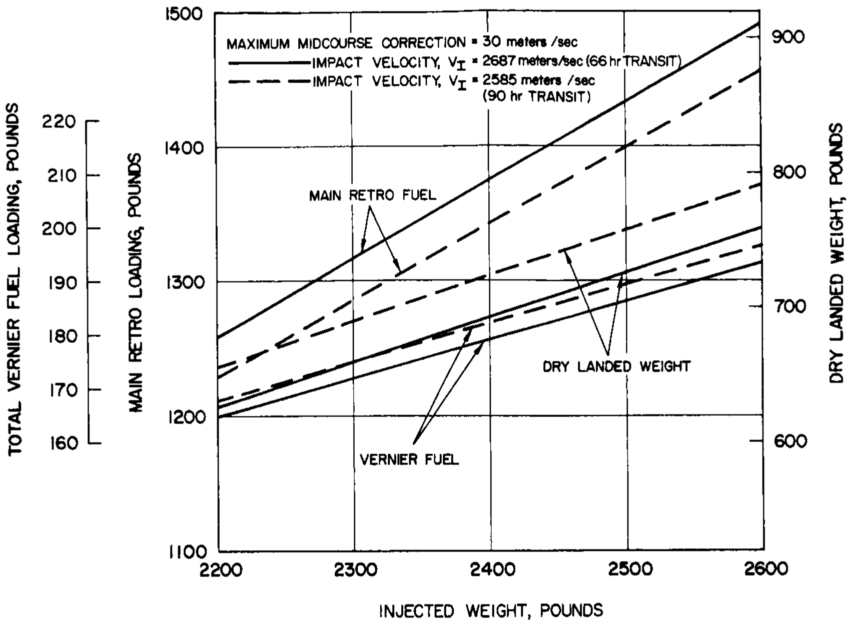


c) Steel Main Retro Case
Aluminum Main Retro
Propellant

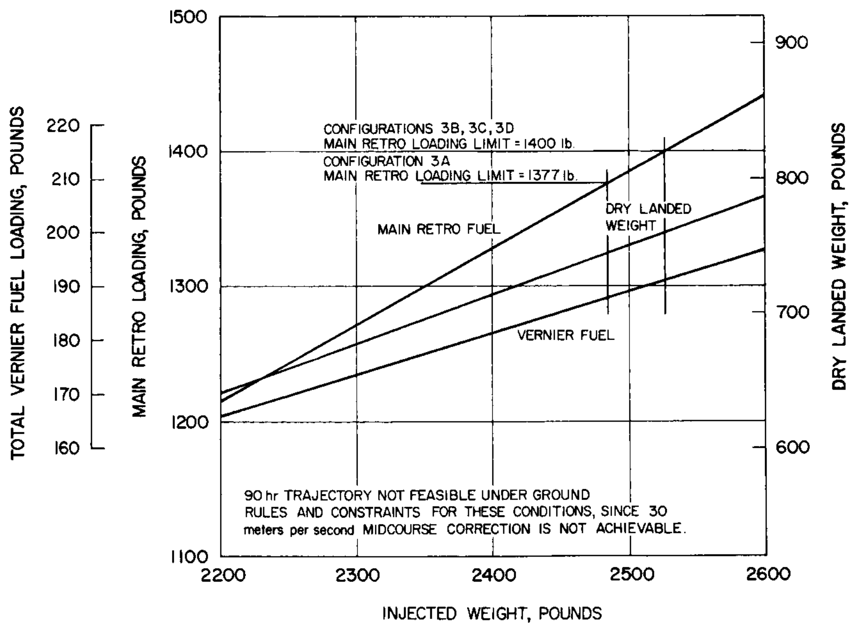


d) Steel Main Retro Case
Beryllium Main Retro
Propellant

Figure 3-10. Dry Landed Weights and Required Propellants
Extended throttle range type vernier engines (9 to 1)

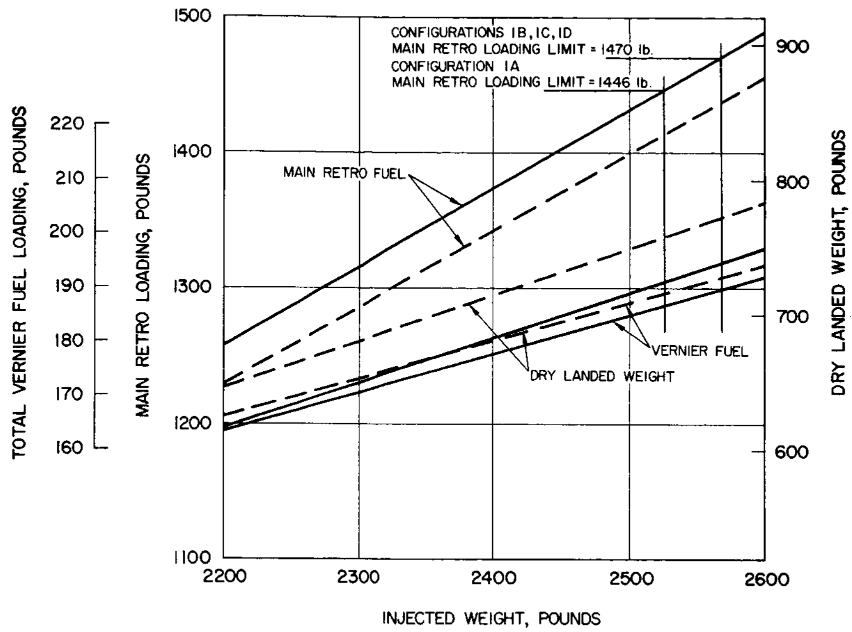


a) Titanium Main Retro Case
Aluminum Main Retro
Propellant

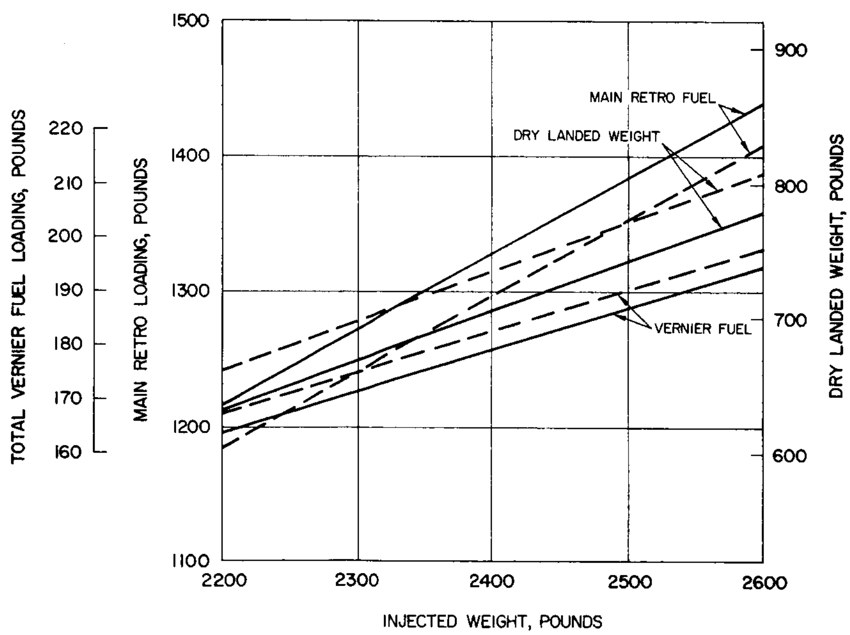


b) Titanium Main Retro Case
Beryllium Main Retro
Propellant

Figure 3-11. Dry Landed Weights and Required Propellants
Restricted throttle range type vernier engines (3.5 to 1)



c) Steel Main Retro Case
Aluminum Main Retro
Propellant



d) Steel Main Retro Case
Beryllium Main Retro
Propellant

Figure 3-11. Dry Landed Weights and Required Propellants
Restricted throttle range type vernier engines (3.5 to 1)

Surveyor A-21A dry landed weight 601.5 pounds

Injected weight change

$$\frac{\partial(\text{main retro weight})}{\partial \text{ dry landed weight}} = 1.66$$

$$\frac{\partial \text{ (vernier fuel)}}{\partial \text{ dry landed weight}} = 0.26 \qquad (\text{total}=1.92)$$

Thus, change in dry landed weight =

$$(2200 - 2150) \times \frac{1}{1 + 1.92} = \qquad +17.1$$

Case weight change

$$\frac{\partial \text{ dry landed weight}}{\partial \text{ (burnout weight)}} = 0.9$$

Change in dry landed weight =
(142.9 - 145.8) x 0.9 = -2.6

Impact velocity decrease

$$\frac{\partial(\text{main retro weight})}{\partial V_I} \cong \frac{1 \text{ pound}}{3 \text{ meters per second}}$$

Change in dry landed weight = (2692 to 2687) x $\frac{1}{3}$ = +1.7

Total calculated dry landed weight	617.7
Dry landed weight	617.5
Discrepancy	0.2 pound (negligible)

LAUNCH WINDOW CONSIDERATIONS

Earlier in this section, the maximum impact velocities (for the two transit times: 66 and 90 hours) occurring on any day during the period in which the Block II Surveyor spacecraft is expected to be operational were used to determine the required propellant loadings, and thus the spacecraft dry landed weights. These two critical maximum velocities were 2687 and 2585 meters per second for the 66- and 90-hour trajectories. However, it is clear that the variation of impact velocities is also of significance. The degree to which the spacecraft design can tolerate the required impact speed variation has direct bearing on the available number of launch opportunities.

TABLE 3-6. COMPARISON OF SURVEYOR BLOCK II CONFIGURATION WITH SURVEYOR BLOCK I, A-21A

	Surveyor Block I, A-21A, February 1964 Design	Surveyor Block II, 66-hour Transit, RTRVES, Steel Main Retro Case, Aluminum Main Retro Propellant
Injected weight, pounds	2150	2200
Maximum midcourse correction, meters per second	30	30
Vernier thrust range, pounds	30-104	30-104
Vernier I_{sp} (mid thrust), seconds	284	284
Main retro case weight, pounds	142.9	145.8
Maximum design impact velocity, meters per second	2692	2687
Dry landed weight	601.5	617.5

Table 3-3 shows that if launch is to be allowable at any time from 1967 through 1969, an impact velocity tolerance of 76 meters per second is necessary for 66-hour trajectories, and 73 meters per second for 90-hour trajectories.

By contrast, the velocity variations which can be tolerated by various alternative Block II Surveyor spacecraft designs, as obtained from the performance calculations, are shown in Table 3-7. Thus, with the present RADVS linear doppler limit at 700 fps, the RTRVES spacecraft designs yield an impact velocity range of only 0 to 14 meters per second, while when equipped with the ETRVES and the present RADVS, the velocity variation capability increases to 24 to 29 meters per second. These compare poorly with the required velocity ranges of about 75 meters per second, and a serious launch window problem may exist.

TABLE 3-7. SPACECRAFT IMPACT VELOCITY VARIATION TOLERANCE

Vernier Engine System	Main Retro Propellant	Titanium Main Retro Case Transit Time		Steel Main Retro Case Transit Time	
		66 hours	90 hours	66 hours	90 hours
RTRVES	Al	11	6	14	8
RTRVES	Be	2	0	4	0
ETRVES	Al	24	28	24	28
ETRVES	Be	26	29	26	29

Note 1: Maximum midcourse maneuver = 30 meters per second
Note 2: Spacecraft injected weight = 2400 pounds
Note 3: Velocity variation shown in meters per second

There are two general approaches to the problem: 1) reducing the required impact velocity variation, and/or 2) making the spacecraft more tolerant of such variations.

Restriction of the Required Impact Velocity Variation

The Surveyor Block II mission will probably be constrained to arrive at the moon when the lighting is proper for certain of the scientific experiments, in particular, for television surveys. It will be assumed that the spacecraft can arrive at the moon only during the 8-day period around full moon. The elimination of the major portion of the synodic month will have a bearing on the variation of impact speed, and it is possible that this could eliminate some of the maxima and minima.

The difference between the synodic and nodical month causes full moon to occur at all lunar declinations over the course of a year. As a result, the 8 days around full moon contain the lunar descending node from February through May, minimum declination from May through August, ascending node from August through November and maximum declination from November through February. The correlation of the maxima and minima in impact speed with lunar declination makes it possible to determine the effect of this lighting constraint upon the variation in speed.

The situation for 66-hour trajectories in 1967 is shown in Figure 3-9. The maximum speed for the year occurs in December at the descending node. However, at this time of the year the descending node occurs near third-quarter moon. The maximum speed for the year, (2654 meters per second) which occurs in the 8-day period around full moon, is in May and June. The minimum speed in February is near full moon. The variation in speed over the year considering this lighting constraint is therefore 2629 to 2654 meters per second.

Similarly, the variation for 66-hour trajectories in 1968 and 1969 is found to be 2620 to 2663 meters per second, and 2611 to 2678 meters per second respectively, for the 8-day period around full moon.

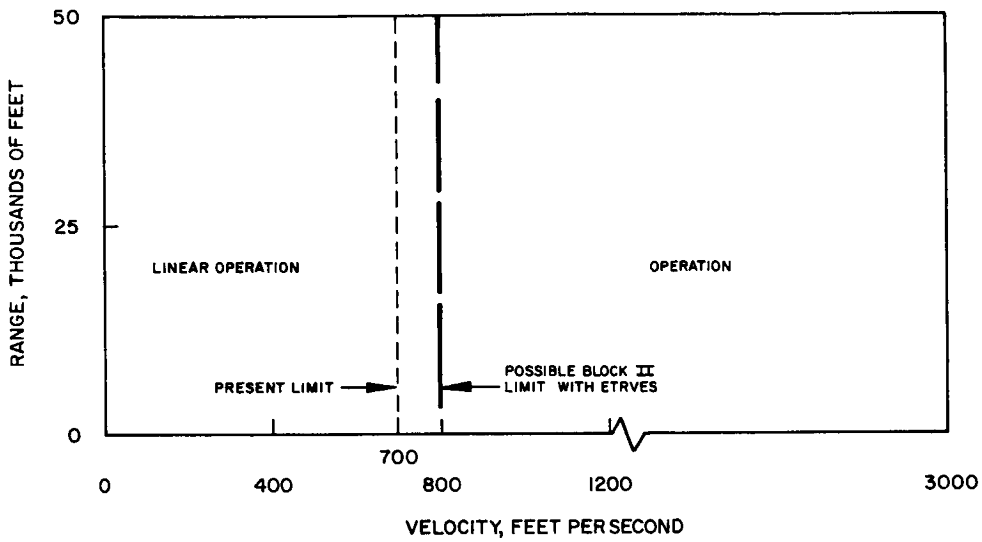
The variation of impact speeds for 90-hour trajectories is not reduced when this lighting constraint is considered. The maxima and minima for all 3 years fall in the 8-day period around full moon. The annual variation in speed ranges from 64 to 69 meters per second for the period 1967 through 1969.

Increasing Velocity Variation Capability

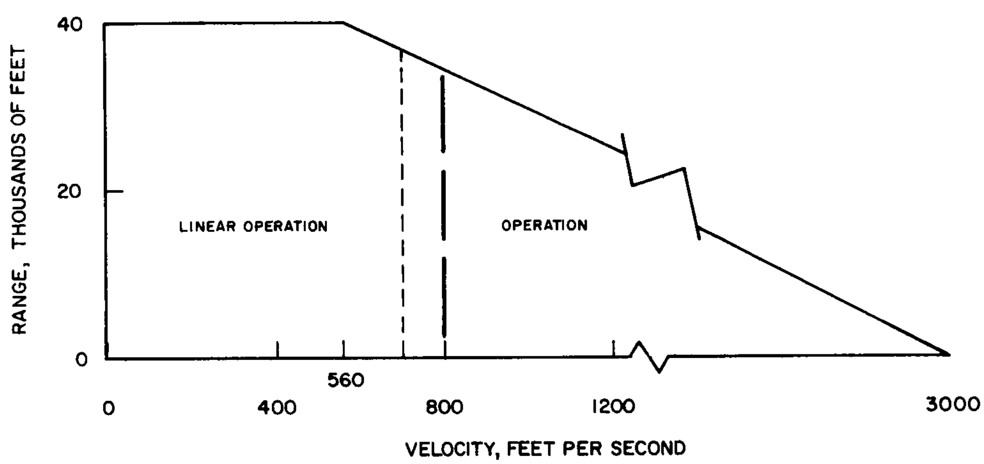
The present specification limit on the upper end of the RADVS range of linear doppler operation is 700 fps. This limits the maximum allowable main retro burnout velocity, and consequently has a direct effect on impact velocity range capability. (The partial derivative of burnout velocity with the respect to impact velocity is approximately 0.95.) If this limit could be changed to 800 fps, the spacecraft employing the ETRVES could realize an additional impact velocity variation tolerance of approximately 32 meters per second. Since the main retro burnout velocity range is altimeter — limited when using the RTRVES, such an increase in RADVS doppler linear range is not useful for increasing the design velocity variation of a spacecraft using the RTRVES.

If the RADVS change were made, system limits would be as shown in Figure 3-12, and it is anticipated that only minor adjustments to present hardware would be required. The upper extremity of the tracking filter frequency search (after retro burnout) would be increased from the present 22 to 24.5 kcps at the cost of a very small decrease in the probability of acquisition. The upper limit of velocity output voltage would increase from 35 volts (700 fps x 0.05 volt/fps) to 40 volts, but the dc amplifiers in the radar have this capability, with perhaps a slight increase in nonlinearity effects. Thus, the RADVS appears to have the growth capability to meet the requirement of linear operation (output voltage a linear function of impact velocity) to 800 fps; however, any increase beyond this point will require hardware redesign, unless compromises are made in sensitivity scale factors and error allowances.

A second way to increase the spacecraft capability to tolerate velocity variation is to attach ballast to the main retro engine. In this way, for a fixed main retro loading the burnout velocity can be raised on



a) Velocity Sensor



b) Altimeter

Figure 3-12. Present, Block I, and Possible Block II Radar Altimeter and Doppler Velocity Sensor low velocity days to higher levels than if ballast was not employed. For example, to increase the burnout velocity range by the same 32 meters per second which the RADVS change would accomplish would require 17 pounds of ballast. However, the ballasting technique is not necessarily feasible. Since the spacecraft is encapsulated atop the centaur during an entire monthly launch period, no greater ballast may be attached to the spacecraft than can be tolerated on the highest velocity day of that launch period. Since velocity variations within a single launch period can be rather high, the amount of ballasting possible, and the degree to which it would help to widen the velocity variation capability of the spacecraft, are questionable. An additional problem associated with ballasting is that injection energy (C_3) is poorly correlated with impact velocity in the time period of interest. Thus, on the minimum impact velocity days, when ballasting is most needed, C_3 may not be at a minimum and the Centaur launch vehicle may not be able to tolerate the additional spacecraft weight that ballasting entails. Within the scope of this study, it was not possible to definitely ascertain the ballasting limits.

Conclusions

Restricted

The RTRVES provides marginal launch window capability under even the best (from an impact velocity variation standpoint) of design alternatives. This condition is that of minimum main retro burnout weight (and minimum dry landed weight), where the altimeter limit is least constraining on the burnout velocity range, i.e., the design using aluminum propellant, steel main retro case, and 66-hour transit time. The resultant impact velocity range is 14 meters per second, which is not expected to provide acceptable launch window capability under constant lunar landing site lighting conditions.

The ETRVES provides a range of 24 to 29 meters per second of velocity variation capability with the present 700 fps RADVS linear doppler limit. The improved RADVS with 800 fps doppler limit, will add 32 meters per second, making a total of 56 to 61-meter per second capability. For 66-hour trajectories, the present radar system can be used for any missions scheduled in 1967. However, the improved system is necessary for 1968 and 1969. The variation of 69 meters per second in impact speed for 1969 can be reduced to the allowable of approximately 60 meters per second by arranging the launch schedule in a judicious manner. This can be done by eliminating several months or possibly only several days in these months. A detailed study would be required to determine possible launch periods. For 90-hour trajectories, the velocity variation is such that the improved RADVS system is needed. It is expected that the variation can also be reduced to the acceptable 60-meter per second range by proper scheduling of the mission. For both 66- and 90-hour trajectories, the ballasting technique is a potential, through questionable, alternative to launch schedule restriction.

4. POWER AND THERMAL CONSIDERATIONS

POWER SYSTEM

The power system considered for Block II Surveyor is similar to the system used on the A-21A series vehicles. This system comprises a planar array, sun tracking solar panel, sealed secondary silver-zinc battery and ancillary electronic circuits to provide conversion, regulation, overload sensing, charge control, and power switching. During periods of daylight, in transit, or on the lunar surface, the solar panel is the prime source of power; the battery supplies the total power load during the night and during transit when the solar panel is eclipsed. The battery also provides power whenever peak demand exceeds the output of the solar panel.

Estimated power requirements for typical 66- and 90-hour transit missions are plotted in Figure 4-1. The peak load during transit is 980 watts, and exists for 4 minutes immediately prior to touchdown.

In addition to the two transit times, three missions are considered: 1) landing and limited survival; 2) 30-day survival, and 3) 90-day survival. Energy source and storage considerations for each of these missions are described in the following paragraphs and summarized in Table 4-1.

The present solar panel area is considered adequate for Block II requirements. The transit and lunar day outputs (Figure 4-2) are net outputs as measured on the unregulated bus after conversion losses are considered. For Block II, the average transit load is assumed to exceed the panel output by 20 watts. The excess energy must be supplied by battery or radioisotope thermoelectric generator (RTG).

The batteries considered in each of the mission types of Table 4-1 are sealed silver-zinc secondary batteries comprised of 14 series-connected cells. Each mission is predicated on the use of two batteries, each capable of completing the particular mission to ensure reliability. The size of the battery considered for each application is based on the excess energy requirements of each type of mission.

TABLE 4-1. ENERGY SOURCE AND STORAGE CONSIDERATIONS FOR BLOCK II MISSIONS

Mission Types	Mission Description	Transit Time, hours	Transit Energy Required, watt-hours	Transit Energy Generated, watt-hours	Transit Energy Deficit, watt-hours	Recharge Time After Transit, hours	Lunar Night Energy Required, watt-hours	Battery Size, watt-hours (Two each-Required)	Total Battery Weight, pounds	Total Weight of Energy Sources, pounds
I	Solar panel-battery system	66	6,400	5,000	1,400	20	0	1,400	45	54
	Limited lunar day survival after landing	90	8,600	6,800	1,800	26	0	1,800	55	64
II	Solar panel-battery system	66	6,400	5,000	1,400	20	~3,500 per night	3,500	93	102
	30-to 90-day survival after landing	90	8,600	6,800	1,800	26	~3,500 per night	3,500	93	102
III	Solar panel-RTG-battery system	66	6,400	6,200	*	Less than 3 hours (not required)	~10,500	500	20	67
	90-day survival after landing	90	8,600	8,450	*	Less than 3 hours (not required)	~10,500	500	20	67

*Insignificant — Battery required to supply peak loads only.

Peak power demand periods occur at postlaunch acquisition, at the midcourse correction maneuver, and during the terminal descent. Since the terminal descent maneuver is normally made with the solar panel eclipsed, the energy used in this phase is not replaced in the batteries until after landing, when repositioning of the solar panel is complete. The nominal time to recharge the batteries after landing is noted in Table 4-1. This period is critical only for landings that occur shortly before the day-night terminator, and when lunar night survival is required.

For mission type II, complete mission redundancy is available. If no battery failure occurs, an additional 3500 watt-hours of energy is available for payload operation during the lunar night.

For the cases studied that use the RTG to provide additional recharge capability and lunar night power, the battery size is determined only by the power demand in excess of the average power provided by the solar panel and RTG during transit and lunar day or by the RTG alone during the lunar night.

The only RTG considered in detail for this study is the SNAP-11. Because of the low probability of spacecraft survival for a 2-year mission (see Section 8), the use of an RTG such as SNAP-9A (see Table 4-2) having a 5-year half-life to provide 2 years of lunar operation is not deemed practicable at this time. The SNAP-11 is being developed by Martin Nuclear Division, Baltimore, Maryland, under AEC — NASA direction for use on Surveyor. The unit is fueled with curium 242 having a half-life of 168 days and furnished by the AEC. The quantity of fuel used provides sufficient power for 120 days after fuel capsule installation into the RTG.

The RTG system considered for Block II Surveyor is essentially the same as that considered earlier for inclusion in the A-21 design, and reported in the Bimonthly Progress Summary SS110, January 1963. In the earlier study a single 15-pound battery was provided; the current study is based on redundant batteries. As reported in the earlier study, the weight of the RTG is 30 pounds, but an additional 8 pounds is associated with substructure, shielding, wiring, and electronic circuitry.

The SNAP-11 RTG design provides a minimum of 15 watts under lunar day environment, 18 watts during transit, and 21 watts during lunar night. The energy balance during transit of the Surveyor Block II RTG power system is shown in Figure 4-3.

A summary of the anticipated weight of each power system is also shown in Table 4-1. The weight of the battery considered for each mission is not a linear function of the battery capacity. The smaller capacity batteries with high discharge rates are not as efficient on a watt-hour per pound basis as the larger capacity batteries. However, a significant weight saving is evident for the RTG system when the short-term energy requirement during transit or after landing on the moon does not exceed 500

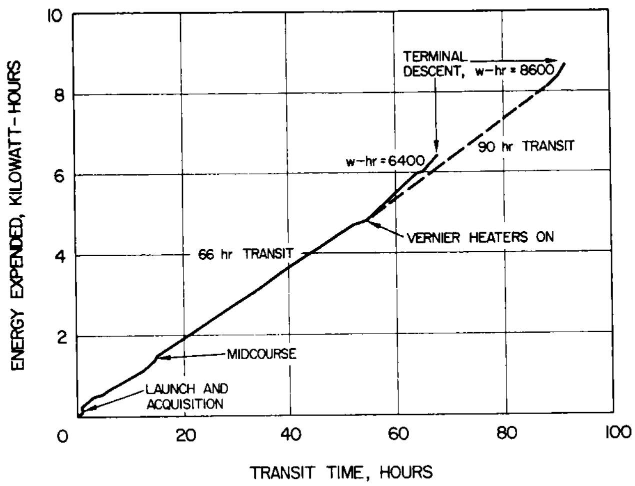


Figure 4-1. Total Transit Energy Requirements

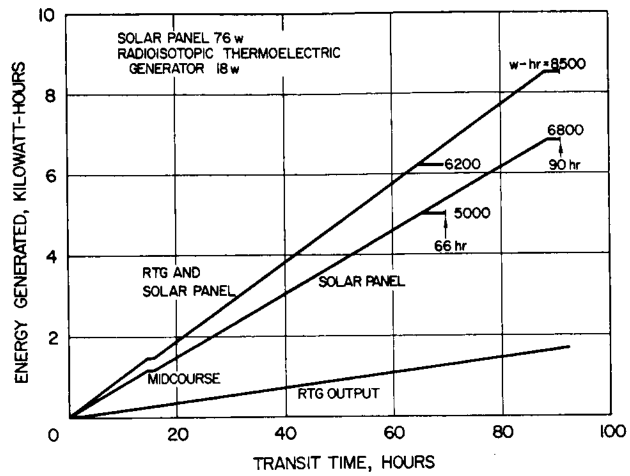


Figure 4-2. Electrical Energy Generated during Transit

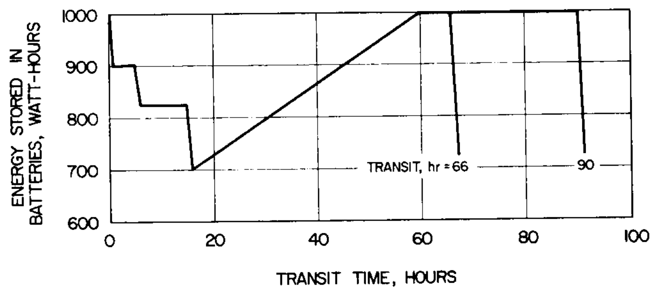


Figure 4-3. RTG Solar Panel Transit Energy Balance

TABLE 4-2. STATUS OF RTG DEVELOPMENT PROGRAMS*

RTG Type	Power, watts	Life, years	Application	Fuel	Status
SNAP-3	2.7	5	Navigation satellite (Navy)	Pu 238	Two launches
SNAP-9A	25	5	Navigation satellite (Navy)	Pu 238	Two launches
SNAP-11	25	1/3	Surveyor	Cm 242	1965 (Electrically heated unit in test March 1964)
SNAP-13	12.5	1/3	Thermionic demon-stration unit	Cm 242	Electrically heated unit in test
IMP	20	1 to 5	Interplanetary monitoring probe	Pu 238	Under development
COMSAT Generator	35	5	MILCOMSAT	Sr 90	Under development

*President L. B. Johnson report to Congress, January 1964.

watt-hours at a maximum discharge rate of about 50 amperes (1000 watts).

For mission type II, it will be necessary to land about 24 hours prior to day-night terminator to allow sufficient time to completely recharge both batteries. Landing for mission type III may occur at any time since recharge requirements are small and can be easily handled by the RTG.

The RTG system offers a weight advantage of approximately 35 pounds relative to the non-RTG system, for both 30- and 90-day survival missions. The RTG system also provides greater reliability and greater mission flexibility. For these reasons it is recommended that the RTG system be incorporated in Block II Surveyor for all missions requiring survival through one or more lunar nights.

THERMAL CONSIDERATIONS

Thermal control provisions for Block II Surveyor are not expected to change appreciably from the A-21A design. However, additional heater power will be required to maintain the vernier fuel and oxidizer tanks at acceptable temperatures if a 90-hour transit is employed.

Although a detail analysis has not been made on the thermal control requirements for a beryllium propellant retro-engine, several general conclusions may be made. Experience gained by analysis and test on the present aluminum propellant retro-engine has confirmed that the temperature gradient which could exist in the grain would be about 50°F for the 90-hour transit case with a predicted minimum temperature of 20°F. The uncertainty of this minimum temperature is ±11°F. Tests conducted on engines conditioned to these temperature gradients have given satisfactory results.

The temperature sensitivity of the Be propellants is from two to five times more severe than Al propellants while the grain density is lower. Both of these factors result in the necessity for increased thermal control of the Be grain. The increased temperature sensitivity would require the incorporation of temperature control provisions to ensure that the temperature gradient would not be greater than 10 to 20°F to assure uniform burning and to maintain predictable action time. Secondly, the decreased density of the grain would indicate a lower coefficient of thermal conductivity within the grain which may tend to cause larger temperature gradients than would exist in the Al propellant grain under the same environment. The net result of these factors would be an addition of about 6 pounds in insulation and heaters, with an additional expenditure of about 10 watts of heater power to reduce the thermal gradient within the grain and to maintain a high bulk mean temperature of the grain. A detailed computer analysis and test program to confirm these predictions is not within the scope of this preliminary study program.

On the A-21A design, to reduce the heat loss from the compartments during the lunar night, all wires entering the compartments not required for lunar surface operation are severed. Low conductivity inserts are used to reduce the loss further in the unsevered wires. These measures reduce the heat loss by about 10 watts per compartment. A brief investigation of radioisotopic heaters was made to replace the severing device. The heater would surround the harness at the compartment exit to act as blocks to the heat loss. This device had been proposed previously for a similar application and used polonium 210 as fuel to provide 5 thermal watts. For Block II Surveyor, using curium 242 as fuel, this device may be practicable as a potential additional weight saving. Further study is necessary to ascertain shielding and protective provisions to determine if the application results in a potential weight saving.

With the RTG, sufficient power is available to eliminate the harness severing device, provided that the low thermal conductance inserts are used in the wires.

5. IMPLICATIONS OF INCREASED WEIGHT ON LANDING GEAR AND STRUCTURE

A study has been made of the changes to the landing gear and spacecraft structure caused by growth in spacecraft injected weight up to approximately 3200 pounds. It is desirable to minimize changes particularly in the external spaceframe geometry and configuration. In this regard, future candidate payloads should be selected and designed to simplify payload/ spacecraft integration.

LANDING GEAR

The major variation which has been examined is the effect of increased landed weight as the injected weight of Surveyor is increased. It has been assumed that payloads will be placed upon the spacecraft so as to maintain the vertical center of gravity location at touchdown within the region of 16 to 19 inches above the leg pivot tube center line. Furthermore, it has been assumed that the radius of gyration about the X- and Y-axes will remain in the range from 28 to 32 inches, the present landing system constraints.

The study is also based on using the same vehicle velocity and attitude design criteria as for A-21A; lateral velocities and incidences within the dispersion ellipse contained by lateral velocities of ±7 fps and incidences of ±10 degrees, and a vertical velocity not to exceed 20 fps.

Previous touchdown dynamics studies having been conducted on a unit mass basis; it is possible to scale up parameters in accordance with touchdown weight and retain the same stability and rigid body inertia load characteristics. The landing gear system parameters for both the shock absorber and crushable block force for various ranges of landed weight are shown in Table 5-1. Each range of landed weights corresponds to a 100-pound variation in injected weight, and includes an allowance for the variation in vernier propellant remaining in the tanks at touchdown.

Although tuning of shock absorber and crushable block parameters will be necessary as a function of injected weight, the present basic shock absorber design would be used to the maximum injected weight possible that

TABLE 5-1. LANDING SYSTEM PARAMETERS

Acceptable Landed Weight, pounds	Center-of-Gravity Limits, inches	Compression Damping Constant lb sec^2/in.^{2}	Spring Constant, lb/in.	Rebound Damping Constant, lb sec^2/in.^{2}	Preload Force at 0° F, pounds	Shock Absorber Load, pounds	Crushable Block Force, pounds
595 to 715	16 to 19	0.375	265	8.00	145	7,300	2070
630 to 750	16 to 19	0.394	278.5	8.40	152	7,670	2170
660 to 780	16 to 19	0.411	290.5	8.77	159	8,000	2270
690 to 810	16 to 19	0.428	302.5	9.13	165.5	8,330	2365
715 to 845	16 to 19	0.444	314.5	9.49	172	8,660	2455
745 to 875	16 to 19	0.4615	326.5	9.85	178.5	8,990	2550
773 to 907	16 to 19	0.4785	338.5	10.23	185	9,320	2645
800 to 940	16 to 19	0.496	350.5	10.58	192	9,660	2740
828 to 970	16 to 19	0.513	363	10.95	198.5	9,990	2830
855 to 1005	16 to 19	0.530	375	11.32	205	10,330	2930
885 to 1035	16 to 19	0.548	387.5	11.68	212	10,670	3025

still permits a 10-percent margin of safety. The present design is being qualified to loads corresponding to previous A-25 levels, resulting in a margin of safety of about 30 percent at the present A-21 load levels. Based on stress analysis and the single maximum load drop test to date, the shock absorber will require strengthening beyond an 8000-pound load, corresponding to a maximum 2430 pounds injected weight with a 3.5:1 thrust ratio vernier, Al propellant/steel case retro, and a 66-hour trajectory; and a maximum of 2200 pounds with a 9:1 thrust ratio vernier, Be/Ti retro, and a 90-hour trajectory.

Tuning of shock absorber parameters would require only minor changes in the unit. As the landed weight is increased, the shock absorber spring constant will be increased by removing a portion of the beryllium copper pressurized tube (see Figure 5-1), thereby reducing the silicone fluid volume and reducing the unit weight. Since the preload force must also be increased as landed weight increases, the helium pressure will be increased, necessitating a thicker support column wall thickness resulting in increased weight. A new metering rod groove profile will also be required for each 100-pound increment in injected weight. The result will be essentially the same shock absorber weight of 3.9 pounds for all units up to the maximum load level of 8000 pounds

For landed weights requiring shock absorber loads reasonably in excess of 8000 pounds, up to 8660 pounds, the unit could be strengthened with only minor modifications. (This would be usable for dry landed weights of up to 745 pounds.) In order to improve column stability, it would be necessary to increase all tubular wall thicknesses so that the column would have an increased section modulus. Only the touchdown simulating drop test and vibration test portions of the type approval test (TAT) should be required to requalify the new unit. Hopefully, completion of the TAT tests presently in progress will show a capability of the present design to the 8660-pound load level, so that no weight increase would be necessary for this condition.

For dry landed weights in excess of 745 pounds, it would be necessary to increase the outside diameter of the unit to gain column buckling strength. Such a new unit would be designed to have a 10-percent margin of safety for the maximum expected dry landed weight condition. Again, tuning of the spring and damping constants would be required for 100-pound increments in injected weight below this maximum value. For the maximum landed weight situation (that of the 9:1 thrust ratio vernier, the Be/Ti retro, and a 90-hour trajectory) the strengthening required for a 2600-pound injected weight having a dry landed weight of 836 pounds would result in a weight of 5.5 pounds per unit with a 10-percent margin of safety.

STRUCTURE

The structural weight depends primarily on design load criteria. As a first approximation, it has been assumed that all design accelerations during boost, retro firing, and lunar touchdown are nominally invariant with injected (or touchdown) weight. This is reasonably true for response to boost vibrations, and will be a scaling requirement in adjusting the shock absorber and crushable block parameters. Although functional requirements such as spacecraft center-of-gravity control may be a factor in sizing structural members in a very few regions, structural integrity was the only consideration in this study.

Table 5-2 shows the weight increase associated with each spaceframe structural member corresponding to a spacecraft dry landed weight change from 580 to 900 pounds. This range corresponds approximately to a total spacecraft weight of 2100 to 3200 pounds using current propulsion characteristics. Structural items are classified by class (i. e., there may be more than one member in a class) and are identified in Figure 5-2. Substructure (component support structure) has not been included in this summary, based on the premise that current basic bus items will not change in weight and substructure weight required for new payload (or basic bus) items will be included in payload weight estimates. Item 22, not shown in Figure 5-2a, consists of tubular braces used to shorten the effective column length of several major spaceframe members.

Figure 5-3 includes two curves showing the rate of structural weight increase versus spacecraft dry landed weight. One curve reflects only spaceframe weight changes while the second curve shows spacecraft plus landing gear changes combined. The structural weights allow for about 100 pounds increase in landed weight for the condition of minimum fuel expenditure. Figure 5-3 also indicates the approximate number of structural items affected at several spacecraft weight plateaus. In Figure 5-3, spaceframe geometry and tube diameters have been assumed unchanged. However, above approximately 800 pounds, there is a weight penalty if the basic spaceframe geometry and tube diameters are maintained unchanged. Some reduction in the structural weight increase shown in Figure 5-3 for a spacecraft weight above approximately 800 pounds would be achieved by modification of basic geometry and tube diameters to maintain an optimum strength to weight ratio. A dry-landed weight of 800 pounds corresponds to an injected weight of 2766 pounds for the case of an Al/steel retro, 3.5 : 1 thrust ratio vernier, and 66-hour trajectory; but only 2503 pounds for the Be/Ti retro, 9 : 1 thrust ratio vernier, and 90-hour trajectory.

TABLE 5-2. ESTIMATED SPACEFRAME WEIGHT INCREASES FOR 3200-POUND SURVEYOR SPACECRAFT

Item Number*	Item	A-21A Weight of Item, pounds	Weight Increase for 900-pounds Dry Landed Weight, pounds
1	Pivot — landing gear	1.32	0
2	Tube — upper main to landing gear	3.12	0
3	Fitting — lower center	2.37	0
4	Fitting — mast upper	0.74	0
5	Tube — column base to main lower	1.92	0.33
6	Fitting — propellant tank	3.60	0.62
7	Tube — column	0.90	0.16
8	Fitting — column base	8.40	1.88
9	Tube — main lower	6.96	1.66
10	Cluster — upper (1, 2 and 3)	0.70	0.19
11	Shock absorbers	11.70	6.30
12	Cluster — lower (1, 2 and 3)	0.19	0.07
13	Tube — mast tripod	1.69	0.63
14	Fitting — retro adapter	2.40	0.94
15	Tube — cluster to main lower	3.60	1.53
16	Tube — mast base	0.74	0.35
17	Fitting — socket landing gear	2.40	1.14
18	Fitting — upper column	3.52	1.84
19	Fitting — mast lower	0.27	0.14
20	Tube — main upper	3.21	1.68
21	Tube — base to landing gear	1.26	0.66
22	Bracing — (bolts included)	5.26	2.75
23	Rivets — bolts, etc.	3.60	1.89
24	Landing gear	26.01	13.60

*See Figure 5-2.

VG 5

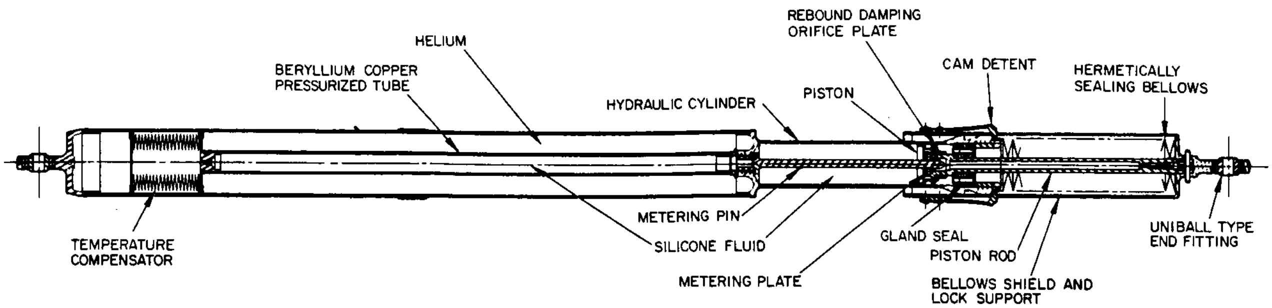
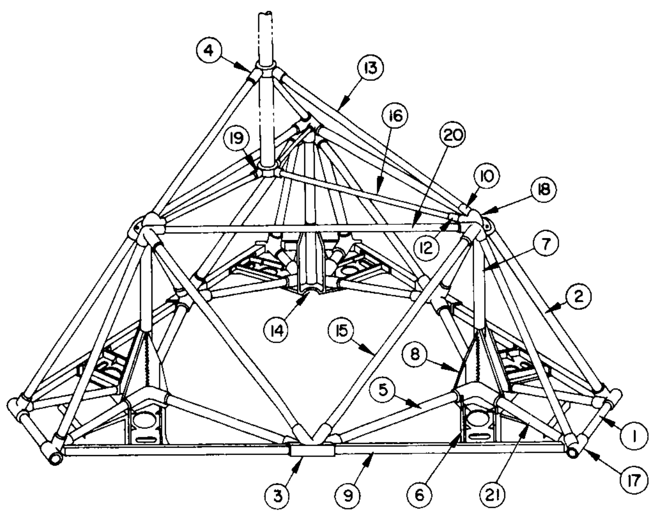
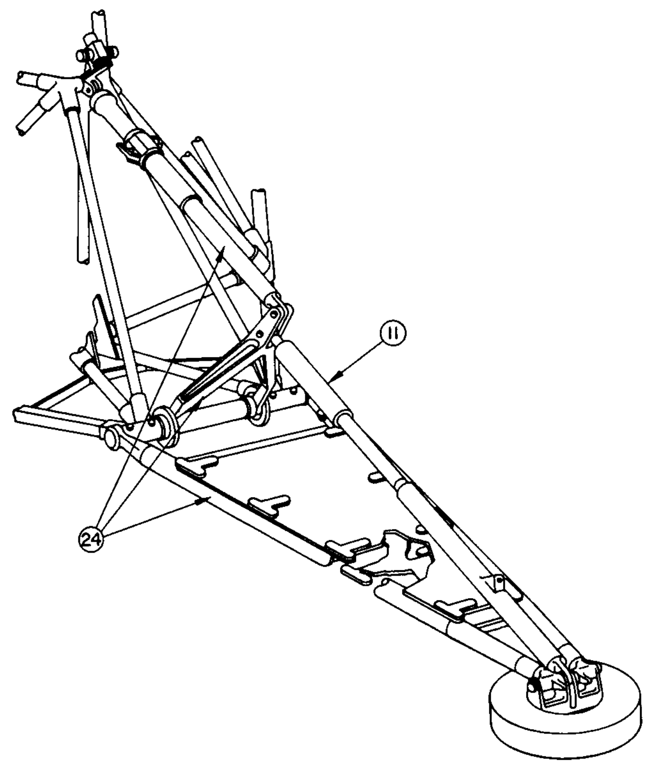


Figure 5-1. Shock Absorber Column Assembly



a) Spaceframe



b) Landing Gear Assembly

Figure 5-2. Spaceframe and Landing Gear Structural Members

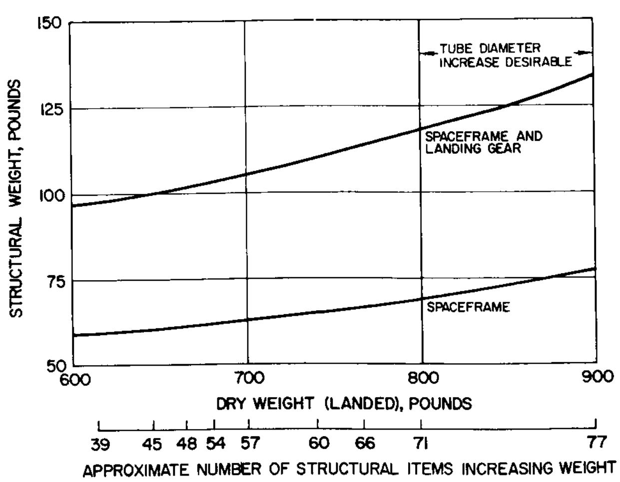


Figure 5-3. Structural Weight Changes as Function of Dry Landed Weight

6. RECOMMENDED IMPROVEMENTS IN BASIC BUS

The ground rule of minimum change in the basic bus design that will have been proven in the A-21/A-21A program restricts the possible improvements to be considered for Block II to those which offer clear advantages in terms of reliability, payload weight capability, or operational flexibility. The use of redundant batteries to improve reliability is recommended in the Reliability section of this report (Section 8), and is incorporated in the power system considerations of Section 4. The RTG power system discussed in Section 4 also enhances reliability and provides operational flexibility. In the following paragraphs, three additional basic bus improvements are discussed.

INTEGRATED SIGNAL PROCESSING

The signal processing subsystem in A-21A Surveyor consists of four units: Central Signal Processor (CSP), Engineering Signal Processor (ESP), Signal Processing Auxiliary (SPA), and Low Data Rate Auxiliary (LDRA). Included in this subsystem are the electronics required for data commutation, signal conditioning, analog-to-digital conversion and the required signal summing prior to RF transmission. There are over six thousand parts in this subsystem and the total subsystem weighs 11.3 pounds.

It is recommended that the four signal processing units be combined into one control item. Thus, weight savings would be realized in total sheet metal, electrical connectors, and intra-unit wiring. In addition, a review of the functional design of this subsystem is expected to reveal means to reduce the total part count, especially with all the electronics in one unit. For example, certain control functions could be combined, eliminating flip-flops and diode gating. Anticipated weight saving is 1.5 pounds; reliability will be improved as a result of a parts count reduction of approximately 500, and fewer inter-unit connections.

RETRO NOZZLE WEIGHT REDUCTION

The recommended retro nozzle improvement would basically involve the substitution of a carbon cloth rosette liner for the current bulk carbon liner in the expansion section aft of the graphite throat insert. The carbon cloth rosette is currently utilized as a backup material for the graphite throat insert and as such has been subjected to more stringent thermal and mechanical loading than would be experienced in the proposed application.

Test results have indicated that carbon cloth rosette does not spall or erode appreciably; whereas the bulk carbon phenolic has demonstrated rather erratic, although not detrimental, performance with regard to spalling. It is believed that the incorporation of the carbon cloth rosette material will eliminate this condition.

Because of the preferential fiber orientation in the carbon cloth and the characteristics of the rosette pattern, the tensile strength of the carbon cloth rosette material is approximately 2.2 times that of the present bulk carbon material. The additional strength available will permit removal of the current exterior fiberglass rosette aft of the nozzle closure and also permit a reduction of 0.080 inch in average wall thickness throughout the expansion cone.

No modifications to the throat area are planned. Similarly the exterior of the submerged portion of the nozzle will remain unchanged. The resulting weight reduction of the nozzle assembly will be 6.5 pounds minimum, assuming the use of aluminum propellant. The nozzle improvement is applicable to either the aluminum or beryllium propellant, but the weight saving can be expected to differ somewhat if the beryllium propellant is used. Advantage of this recommended improvement has not been taken in the calculation of payload weights in Section 7 of this report.

EXTENDABLE MAST

The reinstallation of the extendable mast in Block II will allow approximately 10 to 15 watts additional dissipation capability of the compartment heat-radiation system for extended periods each side of lunar noon. Additionally, the extendable mast may allow improvements in operational flexibility of specific payloads, especially near lunar noon. The addition of the mast extension causes an increase in weight of about 4 pounds.

The extendable mast is not recommended for all Block II Surveyor missions, but is an alternate to be considered for each mission depending on payload requirements, and permits a tradeoff between operational flexibility and 4 pounds of payload weight.

7. PAYLOAD CAPABILITY

SPACECRAFT DESIGN MODIFICATIONS TO ACCOMMODATE INCREASED PAYLOAD WEIGHT

During this study a primary objective has been to maximize payload weight while causing the least amount of change in the basic spacecraft design. Several different methods have been employed in approaching this objective and are presented in this discussion. The methods used to maximize payload weight are summarized below.

Improved Propulsion Performance

Changes in this category include use of a beryllium propellant to provide higher total impulse for the main retro engine without a case size change; use of a titanium case rather than a steel case;* use of an improved lighter weight nozzle for the main retro engine;* use of an extended main retro nozzle (with attendant AMR antenna design change) to permit loading of additional propellant into the existing case; and use of vernier engines with an extended throttle range.

Removal of Basic Bus Elements

Items in this category include deletion of the high gain antenna, the antenna solar panel positioner, the supporting mast structure and cables and connectors for these items when not required by the particular mission. The removal of approach television camera 4 and the television auxiliary together with their cables and supporting bracketry, are also included.

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*Weight reductions in the propulsion system are considered as propulsion improvements rather than basic bus weight reductions since weight reductions in the main retro engine or other expendable items cannot be traded off directly, pound-for-pound, for payload weight. Overall loaded weight I_{sp} is improved by such reductions.

Repackaging or Replacement of Basic Bus Elements to Save Weight

Items in this category include the repackaging of the central signal processor, engineering signal processor, signal processor auxiliary, and the low data rate auxiliary into one unit, as discussed in Section 6 of this report. Also in this category is the substitution of a fixed triangular solar panel for the movable solar panel of the current A-21A design (in conjunction with the deletion of the high gain planar array antenna, mast and positioner).

Use of 90-hour Trajectory

The use of a 90-hour rather than a 66-hour trajectory does not significantly change the basic design of the spacecraft but offers an appreciable percentage change in the payload weight capability. The design changes required for a 90-hour trajectory are limited to power system modifications to provide sufficient capacity to handle the additional 24 hours of transit.

DISCUSSION OF MAIN RETRO ENGINE MODIFICATIONS

In considering modifications to the main retro propulsion system one of the study ground rules required that changes be limited to those which could be readily adapted to the existing spacecraft basic bus without significant change. This limitation together with the ground rule of avoiding Centaur launch vehicle changes prevented the consideration of main retro designs 8 and 9 involving the addition to the main retro of a cylindrical center section, as discussed in Section 3. All main retro engine changes considered are therefore limited to:

1) Changes in propellant to achieve greater total impulse

2) Changes in case or nozzle material to achieve a more favorable overall loaded weight I_{sp}

3) Adding more propellant to the existing case to achieve greater total impulse

The use of beryllium propellant falls into category 1, while the use of titanium as a case material or use of a carbon cloth rosette liner in the nozzle expansion section are in category 2. A number of different propellant loading schemes were considered under category 3. The most effective change involves moving the main retro nozzle out of the case to provide additional volume for propellant. Since it is desired to maintain the same propulsive action time with increased total impulse, the engine operating pressure is increased which, for the most attractive configuration, permits a slight scaling down of the nozzle dimensions while maintaining the same nozzle expansion ratio. This main retro design 6 (shown in Table 3-4) was used for fourteen of the spacecraft configurations presented in this section.

DESIGN MODIFICATIONS TO ALTITUDE MARKING RADAR (AMR)

One of the basic ground rules governing this study has been to avoid changes in the Spacecraft/Centaur interconnect structure or the Centaur Shroud. However, to maximize the amount of propellant which can be loaded into the main retro case without altering its dimensions, it is necessary to move the entire nozzle assembly out of the case by several inches. It can be seen in Figure 7-1 that such a nozzle extension cannot be accomplished without a change to either the AMR or to the Centaur launch vehicle itself. Accordingly, the AMR antenna design has been studied to determine how it can best be modified to provide additional nozzle clearance.

Three new AMR antenna design configurations have been examined to determine feasibility. The first two schemes considered are modifications of the existing AMR antenna design while the third is an entirely new planar array antenna. The basic concept of scheme 1 is illustrated in Figure 7-2. The antenna feed is equipped with a rotating microwave joint permitting it to be folded to one side in a stowed position prior to spacecraft separation from the Centaur vehicle. Sometime after spacecraft separation from the Centaur, a command must be sent to the spacecraft causing actuation of a pinpuller located near the base of the feed on the back side of the AMR dish. This action releases a latch which permits the feed assembly to be erected into operating position by means of a spring drive. With the feed in its erected position, the slot in the dish near the base of the feed is closed simultaneously by a parabolic section which rotates into place with the feed.

The design concept of scheme 2 is illustrated in Figure 7-3. In this design, instead of rotating, the feed is retracted straight back in the stowed position to a point where its tip is even with the edge of the dish. The feed is held in the stowed position by a latch until released by a pinpuller on command from earth. Springs are used to drive the feed from the stowed to the extended position. Scheme 2 offers an advantage over scheme 1 because the gain and sidelobe specifications will be more easily maintained once the feed has erected to the proper position. During the development of the present AMR antenna it was determined experimentally that this feed/dish combination with its relatively low sidelobe levels is particularly sensitive to dish irregularity in the vicinity of the feed base. Because of this it is believed that the closure of the dish slot opening on scheme 1 would have to be accomplished with relatively high precision if antenna characteristics are not to be degraded.

Both scheme 1 and scheme 2 are superficially attractive because they are based on the modification of an existing design rather than on a new design. There are, however, a number of mechanical, functional and test disadvantages to either of these designs which have led to the further examination of a planar array design. The task of designing a rotating or retracting feed, while conceptually simple, may be relatively difficult to accomplish with actual hardware. For AMR antenna specifications to be maintained it is required that the position of the end of the feed structure be maintained within ±0.01 inch of its design center. The surface of the dish near the base of the feed must conform to a perfect paraboloid within ±0.005 inch. These requirements, which are difficult to maintain with the present design, would be even more difficult with a collapsible feed structure. From an operational and functional point of view the success of the mission is entirely dependent on receipt and proper execution of the command to erect the AMR feed. To adequately demonstrate the functional reliability of a collapsible feed system, an expensive series of repetitious tests on a number of units would have to be performed, similar to those conducted on the landing gear and omnidirectional antenna boom assemblies. It is expected that such tests would be more difficult and expensive because of the requirement to make precise mechanical and/or microwave measurements of the results of each test. In actual space flight operations it would probably be necessary to telemeter the proper erection of the AMR antenna feed, adding to system complexity. Such telemetry data, if improper erection is indicated, might permit the choice of a different lunar landing spot to achieve a near perpendicular touchdown, minimizing the effects of poor sidelobe performance. Both scheme 1 and scheme 2 will require rearrangement, and possible repackaging of units mounted on the back of the AMR dish, to provide room for the required mechanism.

Figure 7-4 shows the AMR with the proposed planar array antenna. The principal advantages of this configuration are as follows:

1) Unlike schemes 1 and 2 no rearrangement or repackaging of existing AMR components is required because the array is fed in the center in the same manner as the previous dish feed. This also permits existing tooling for subassembly mounting provisions to be used.

2) There are no moving mechanical elements which must be actuated after final functional test.

3) Since the planar array is a flat plate assembly this design permits the nozzle to be extended approximately 1.4 inches beyond the corresponding position using a folding feed system in a parabolic dish.

Figure 7-1 shows the Surveyor/Centaur clearance envelope with modified AMR antenna. This figure shows how the use of the planar array antenna on the AMR will permit extension of the main retro nozzle while still maintaining adequate Centaur clearance. With this arrangement it is possible to lengthen the main retro nozzle by 5.4 inches. It is understood that on future Centaur configurations the hydrogen vent is no longer located in a position that would interfere with this nozzle extension; if this is not correct, the vent must be relocated.

Although the planar array antenna concept of scheme 3 permits the remainder of the AMR to be used without change it does require the development of an entirely new antenna. Preliminary calculations have indicated that the gain of a circular 30-inch planar array antenna would be 34.3 db ±0.3 db. This figure is based on a conservative estimate of the effects of manufacturing tolerances while maintaining sidelobe levels at least 29 db below the main beam. Although the mechanical and thermal environment to be encountered by the AMR antenna is not as severe as those specified for the main spacecraft planar array antenna, the same fabrication techniques would be used in the design of the flat plate array itself. The actual design would be easier to fabricate because of the smaller size and the use of linear rather than circular polarization.

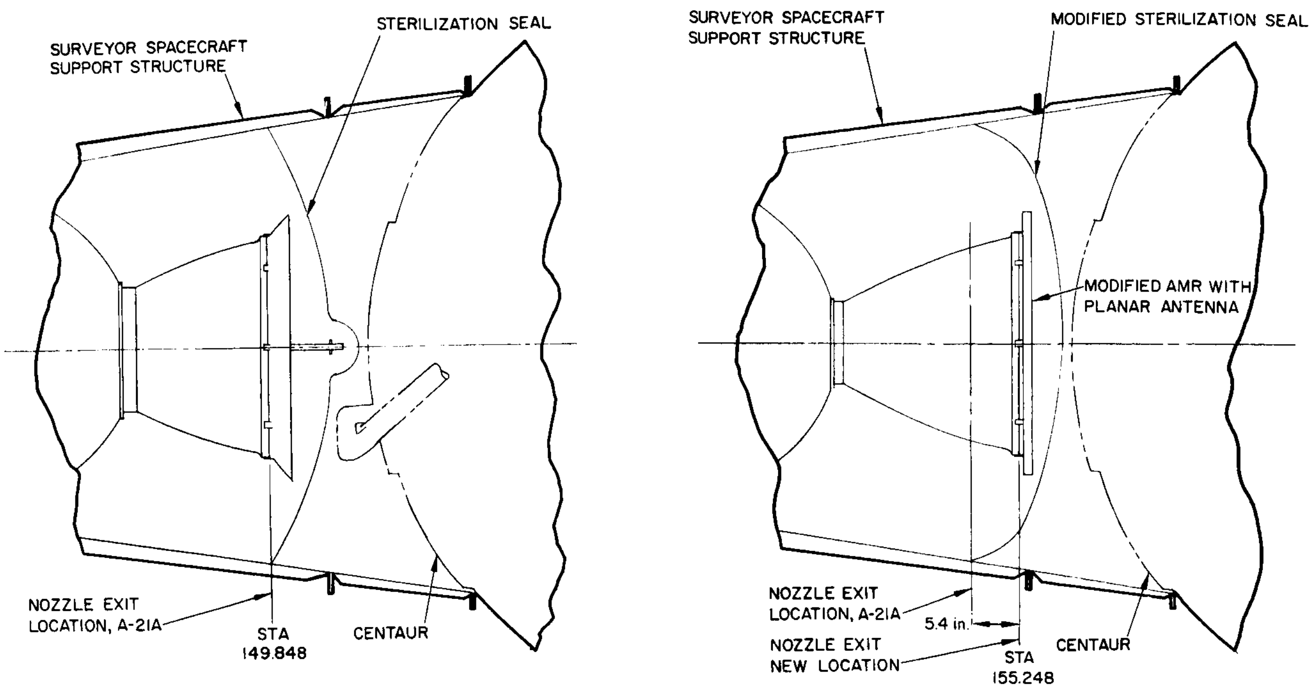
Figures 7-5 and 7-6 illustrate prototype round planar array antennas of 26-inch and 40-inch diameters respectively. These antennas which operate in the same X-band frequency region as the proposed AMR antenna have successfully met their gain and sidelobe specifications. The design and fabrication task for the proposed AMR antenna would be simpler than that performed for the antennas illustrated because neither monopulse nor broadband operation is required.

The planar array antenna of scheme 3 is recommended over either the folding or retracting antenna feed modifications of schemes 1 and 2, for the following reasons:

1) Once a planar array antenna is designed, fabricated, and passed through type approval tests it becomes a proven element not dependent on remotely actuated precise mechanical action.

2) Type approval testing of the planar array antenna is expected to be substantially less lengthy and less expensive than that of a modified folding feed antenna.

3) The degree of risk involved in being able to design and produce a suitable planar array antenna having the required mechanical and electrical characteristics is considered to be substantially less than that associated with the development of a folding feed antenna.



a) A-21A Design

b) Nozzle Extension with Planar Array Antenna

Figure 7-1. Surveyor Spacecraft/Centaur Clearance Envelope

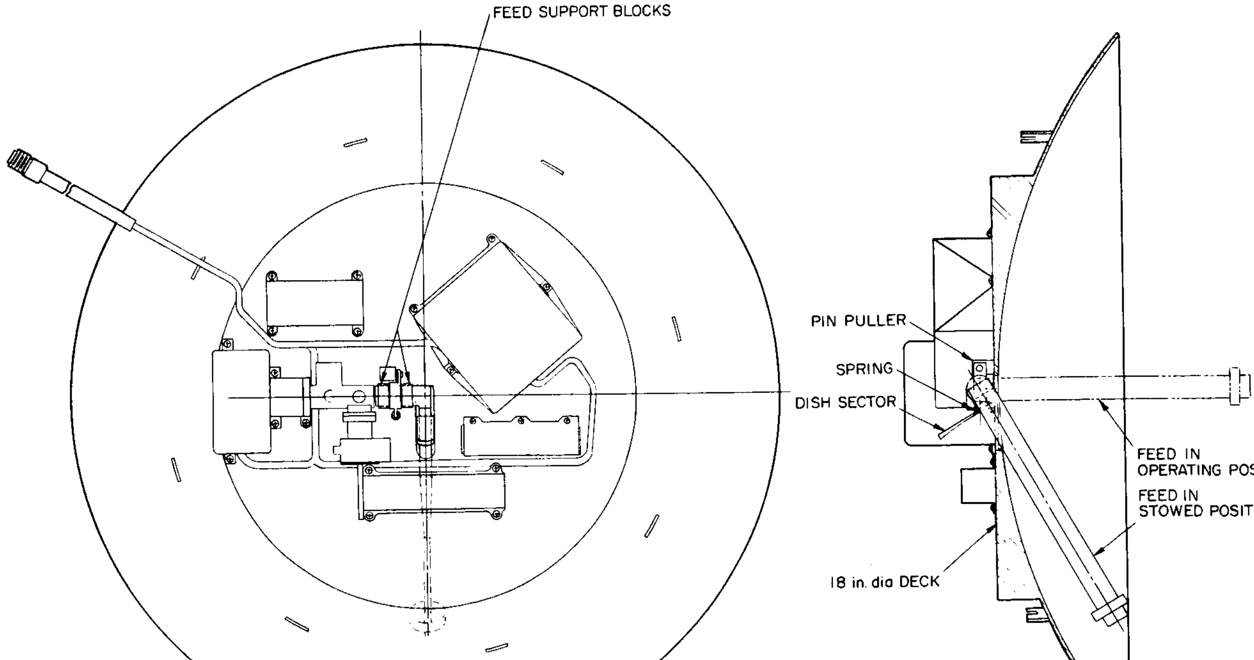


Figure 7-2. Altitude Marking Radar with Proposed Folding Feed Scheme 1

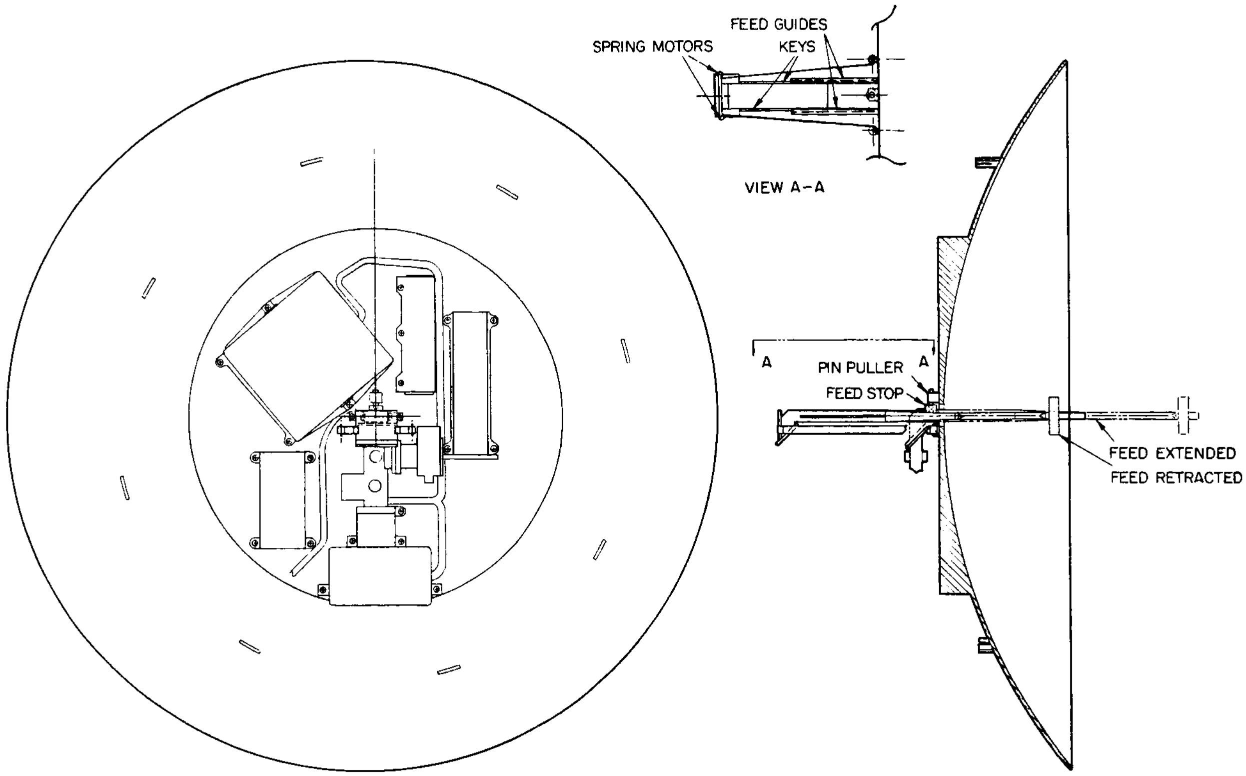


Figure 7-3. Altitude Marking Radar with Proposed Retracting Feed Scheme 2

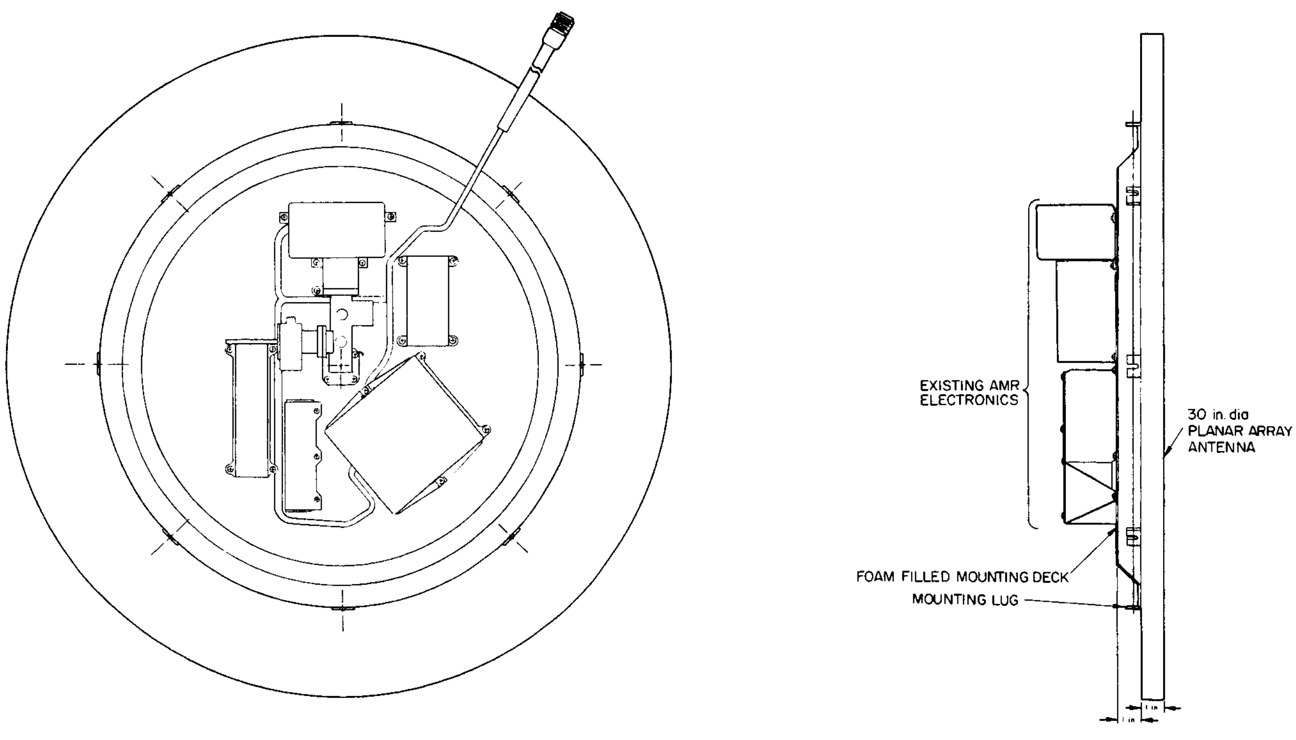


Figure 7-4. Altitude Marking Radar with Proposed Planar Array Antenna Scheme 3

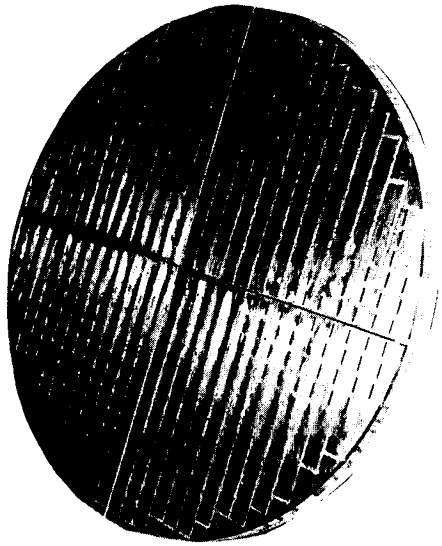


Figure 7-5. 26-Inch-Diameter Planar Array Antenna

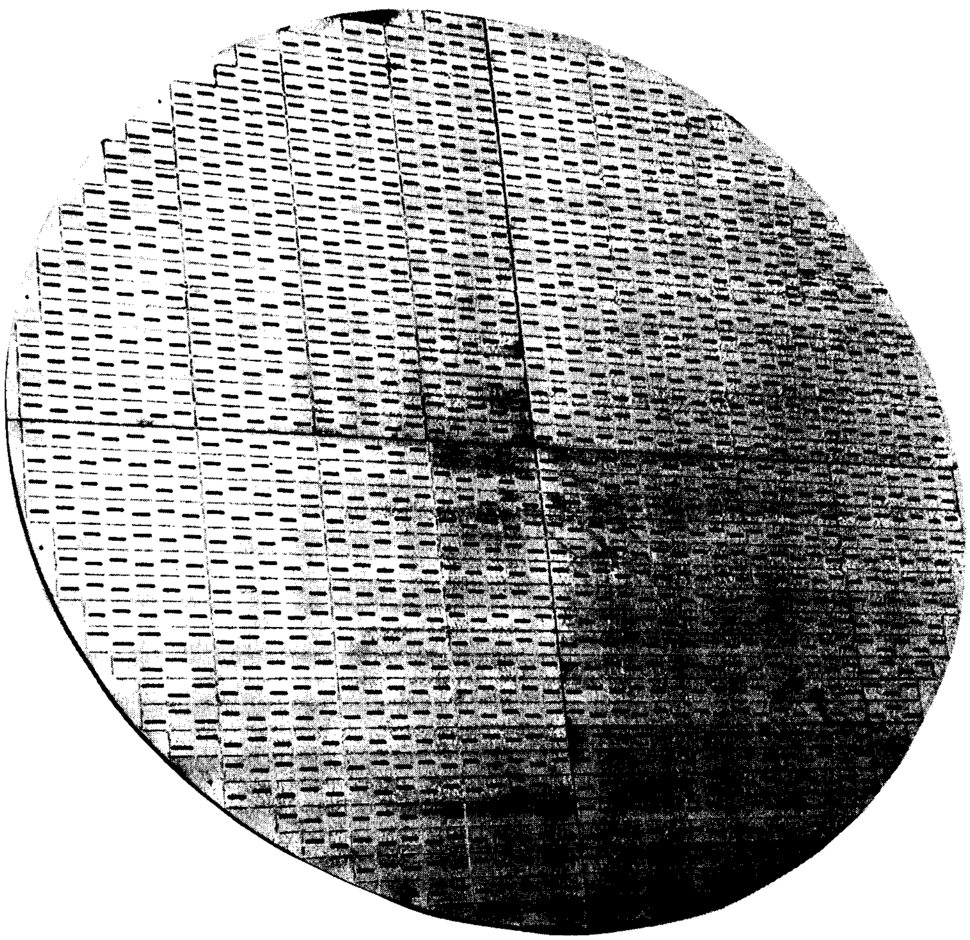


Figure 7-6. 40-Inch-Diameter Planar Array Antenna

4) The costs in development time and funding to obtain a reliable working model AMR with planar array antenna are expected to be no more, and quite possibly less, than those associated with the design, fabrication, and test of a repackaged AMR with collapsible feed.

VERNIER ENGINE PROPELLANT TANK MODIFICATIONS

Missions requiring more vernier propellant than that presently required for A-21A will necessitate relocating the larger vernier propellant tanks outboard a few inches. Structural members above and adjacent to the propellant tanks prevent growth of the tanks in their present location. Repositioning the tanks will make redesign of the spaceframe geometry in the vicinity of the tanks unnecessary. Figure 7-7 illustrates the vernier tank positions for both A-21A and Block II missions. Propellant loadings to about 260 pounds may be accomplished with this modification.

The present vernier propellant tank and RADVS supports require modification to accommodate the new tanks; however, these represent nominal changes to bracket-type hardware. Only three such items are affected.

FIXED SOLAR PANEL SPACECRAFT CONFIGURATION

For missions where the primary purpose of the Surveyor spacecraft is to softland a payload at a given location on the moon without the requirement to provide wideband telemetry or an extended period of spacecraft survival, the use of a fixed solar panel without a high gain antenna offers a substantial saving of about 38 pounds in basic bus weight. A spacecraft configuration employing a fixed solar panel without a high gain antenna is illustrated in Figure 7-8. The solar panel is mounted perpendicular to the spacecraft roll (Z) axis and consequently would receive full normal illumination during the coast phase of transit. The area has been increased to 10 square feet (compared to the 9 square feet of the movable panel) to compensate for increased operating temperature. For this type of mission it is assumed that no requirement exists for TV or other wide bandwidth communication. It is also assumed that two redundant medium-size (1400 watt-hour) batteries would be employed to replace the one large battery now used on A-21A, as described in Section 4. Should one of these batteries fail during transit, the remaining battery would provide sufficient power to permit soft landing and verification of the condition of spacecraft and payload immediately after landing. It would also provide for the commanded deployment of the payload and telemetry of its separation from the basic bus. If neither of the two batteries should fail during transit, it is likely that spacecraft survival could range from 2 hours up to several days, depending on landing site location and solar angle at touchdown. During this time the spacecraft could provide two-way telecommunication with frequency of transmission and bandwidth (e.g., high or low power transmitter operation) determined primarily by the relative solar angle with respect to the fixed solar panel and the thermally controlled compartments. Power generation for spacecraft operation and battery charging could continue at reduced efficiency even without the capability to track the sun with a movable solar array.

ELIMINATION OF APPROACH TV SYSTEM

The deletion of the approach television system from the basic bus saves approximately 11 pounds. Approach TV is not believed to be a requirement for Block II missions, and the decision to eliminate this item is consistent with the objective of maximizing payload capability. If approach television coverage is required by the particular payload being landed, it is logical to allocate the weight of the television system to the payload itself.

PAYLOAD/BASIC BUS WEIGHT TRADEOFF SUMMARY

Items that fall into the category of expendables (main retro engine, AMR, vernier propellant, etc.) that affect dry landed weight cannot be traded off directly pound for pound for payload weight. However, for a given dry landed weight* it is practical to make design tradeoffs in weight allocations between the spacecraft basic bus and the payload. It is useful to summarize the tradeoffs involved between payload weight, basic bus weight, and overall spacecraft performance. A presentation of these factors for various basic bus design modifications considered is shown in Table 7-1. All of the design modifications shown have been discussed previously.

MAXIMUM PAYLOAD WEIGHTS FOR BLOCK II

Figure 7-9 summarizes the results of the study in terms of maximum spacecraft payload capability and total injected weight for 18 different configurations. The current A-21A design is included for reference purposes. All of the configurations presented fall into two main categories: main retro engine with aluminum propellant and steel case, or main retro engine with beryllium propellant and titanium case. Examination of configurations employing the present proven aluminum propellant with the existing steel case is logical in that it represents an extrapolation of the current main retro design. The association of the higher performance beryllium propellant

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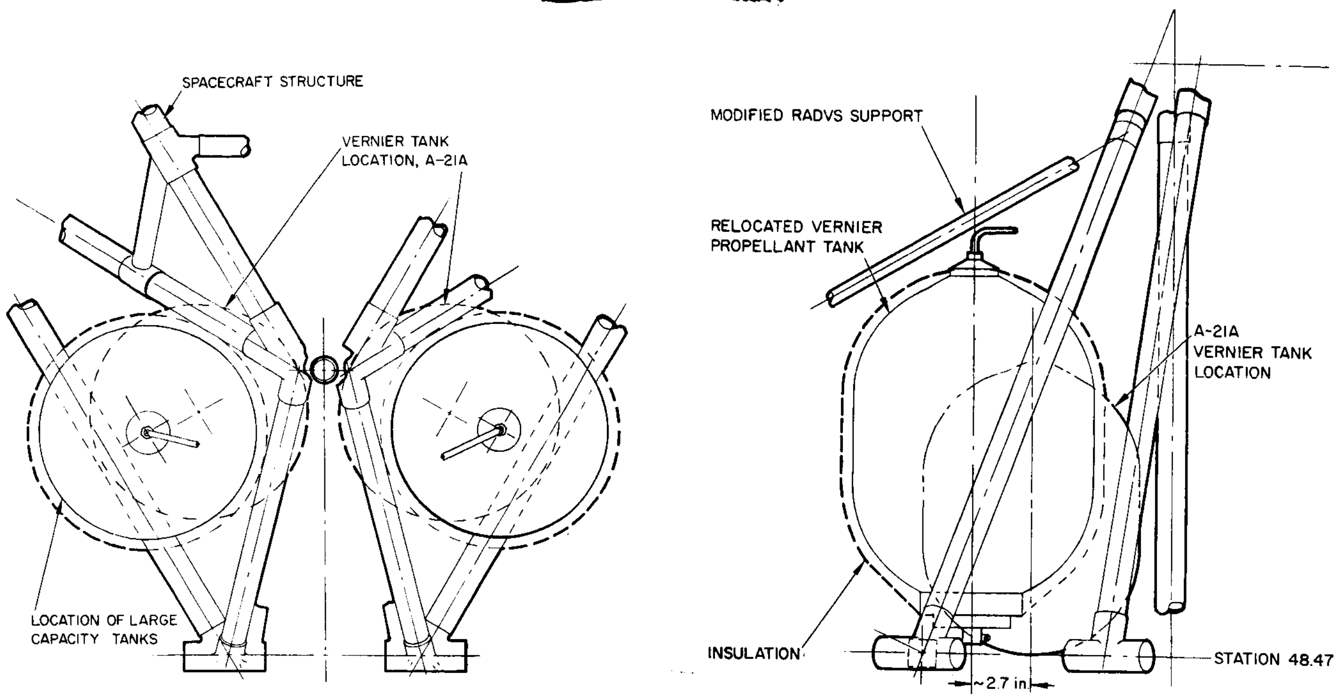
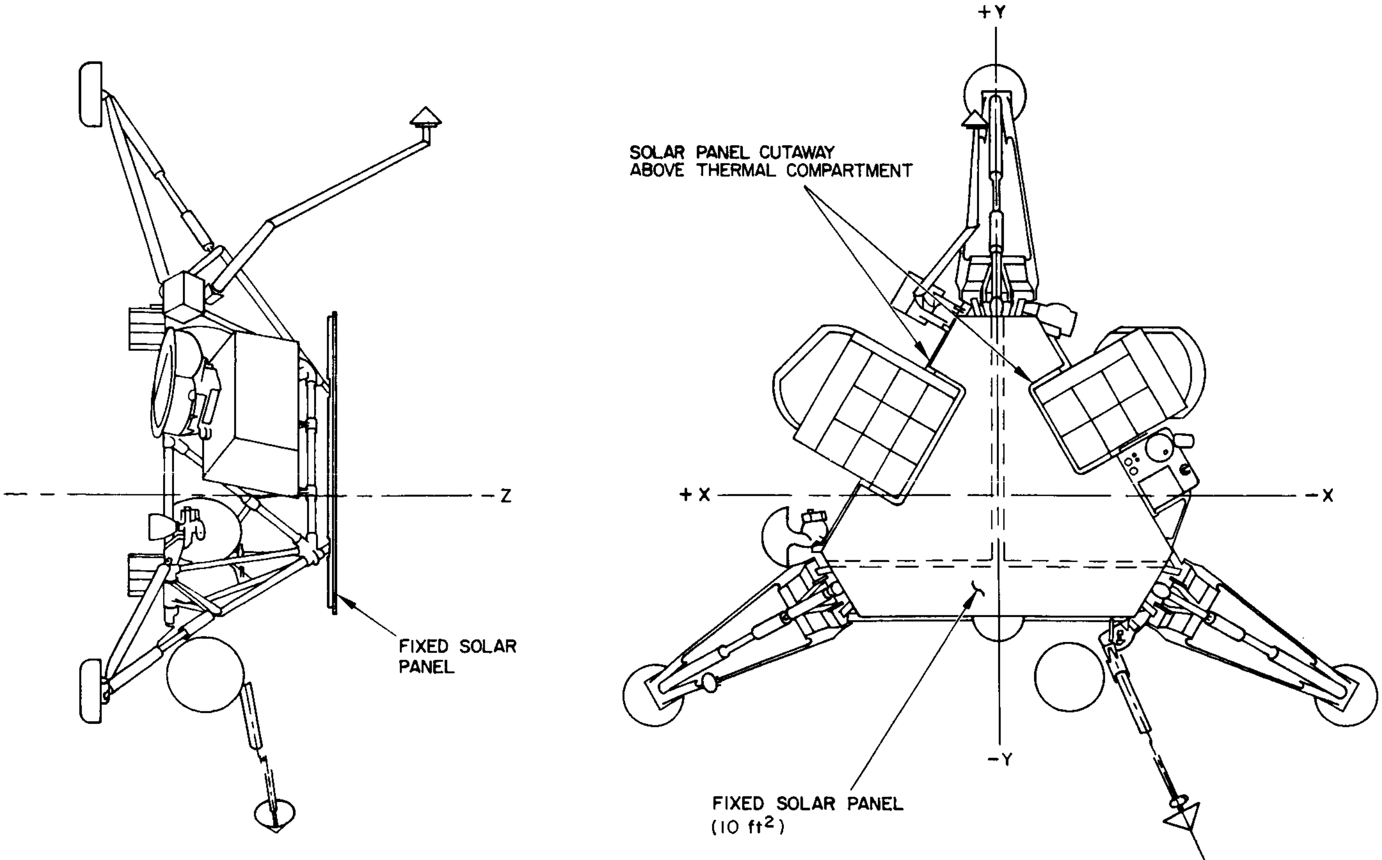
*See page 7-16 for a more complete definition of expendable and dry landed weights.



a) Top View b) Side View

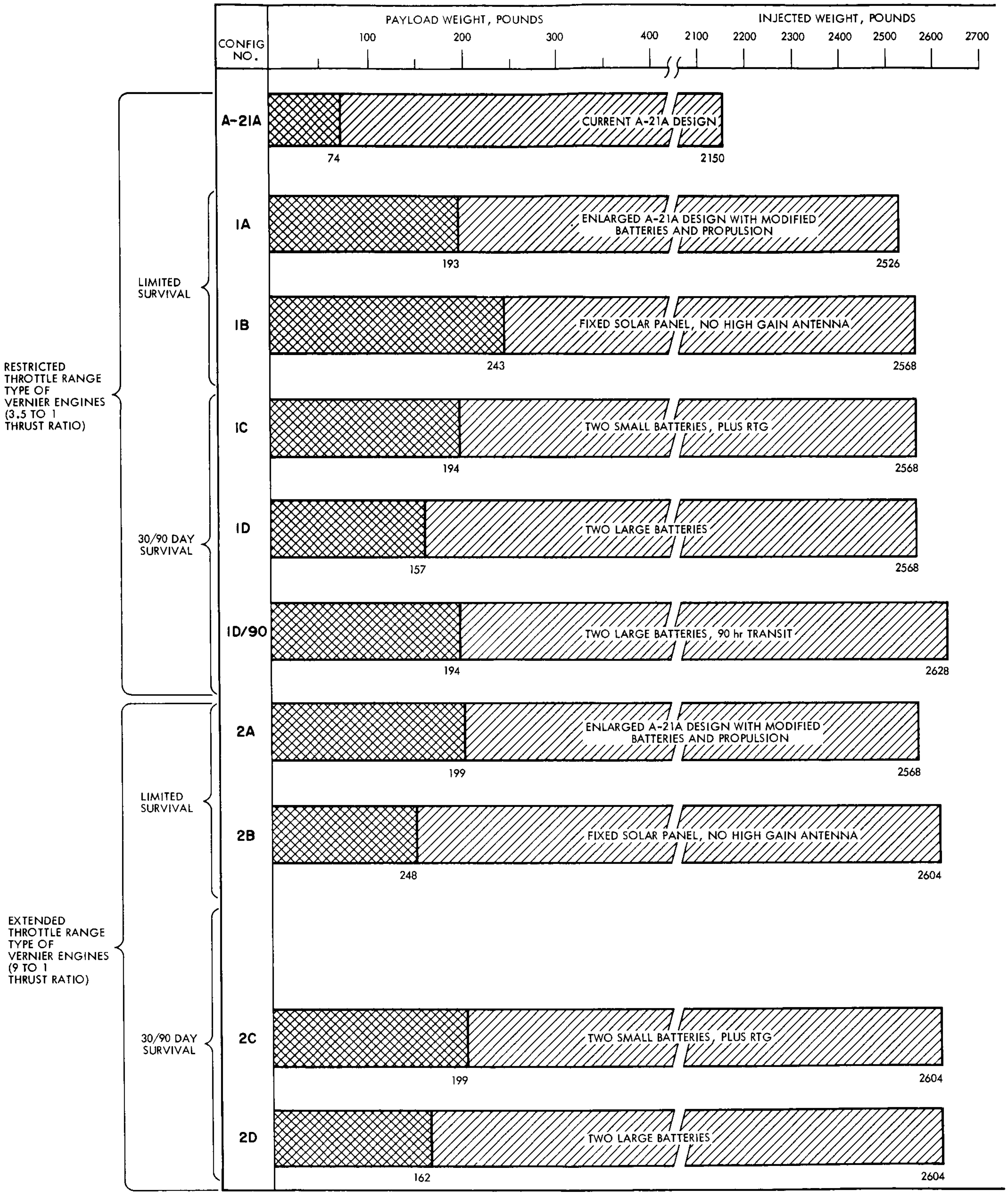
Figure 7-7. Relocation of Vernier Propellant Tanks for Increased Capacity



a) Side View b) Top View

Figure 7-8. Spacecraft Configuration with Fixed Solar Panel

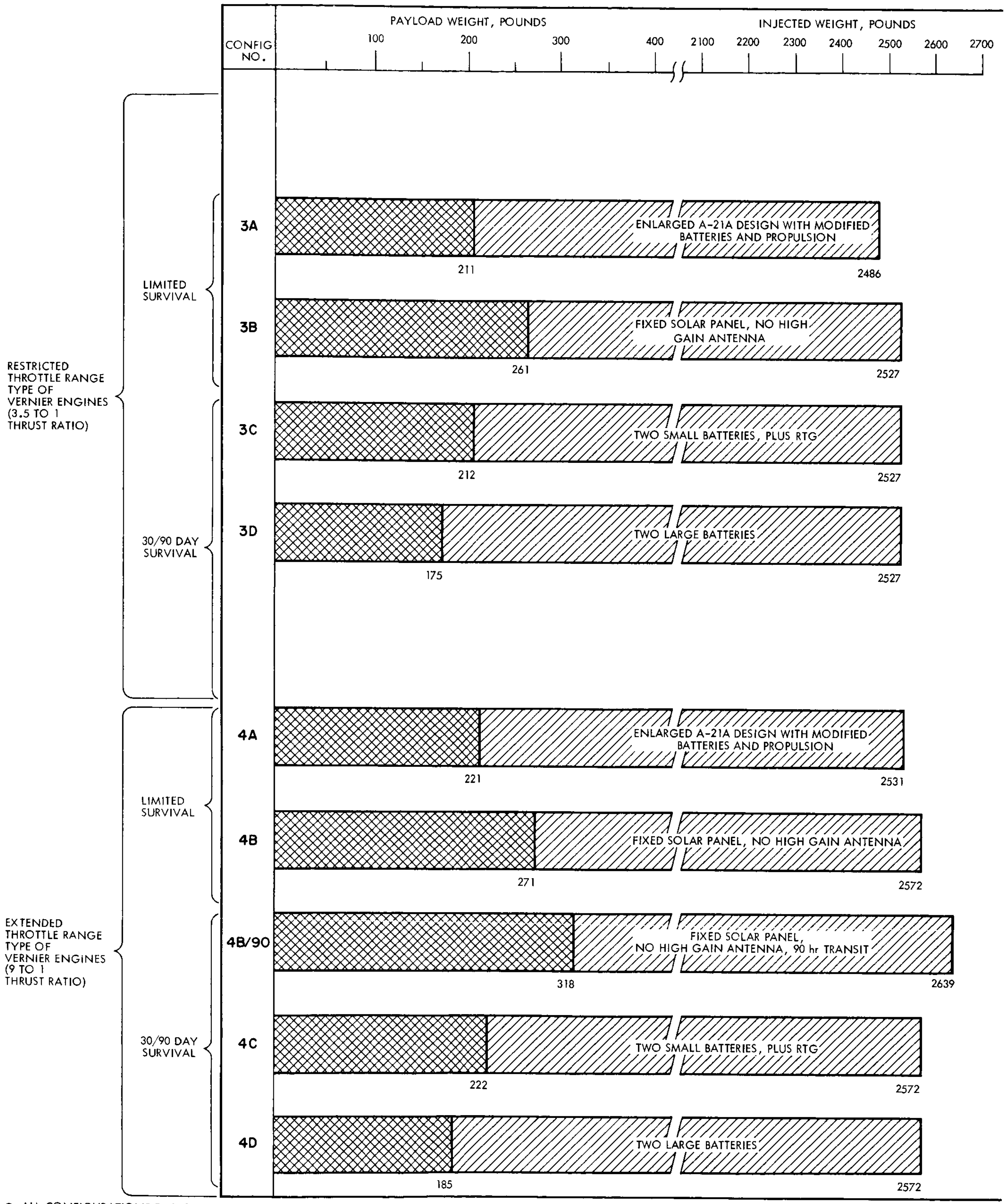
MAIN RETRO WITH ALUMINUM PROPELLANT, STEEL CASE



- ALL CONFIGURATIONS ARE FOR 66 HOUR TRANSIT EXCEPT NUMBERS 1D/90 AND 4B/90
- ALL CONFIGURATIONS EMPLOY EXTENDED MAIN RETRO NOZZLE AND NEW AMR ANTENNA EXCEPT NUMBERS 1A, 2A, 3A, AND 4A

Figure 7-9. Maximum Payload and Injected Weight for Each Spacecraft Configuration Studied

MAIN RETRO WITH BERYLLIUM PROPELLANT, TITANIUM CASE



- ALL CONFIGURATIONS EMPLOY HIGH GAIN ANTENNA AND MOVABLE SOLAR PANEL EXCEPT NUMBERS 1B, 2B, 3B, 4B, AND 4B/90.
- REFER TO TABLE 7-2 FOR DETAILED DEFINITION OF EACH CONFIGURATION

Figure 7-9. Maximum Payload and Injected Weight for Each Spacecraft Configuration Studies

TABLE 7-1. BASIC BUS/PAYLOAD WEIGHT TRADEOFFS FOR DESIGN MODIFICATIONS CONSIDERED

Item Number	Description of Basic Bus Design Modification (Compared with Current A-21A Design)		Basic Bus Weight Change, pounds: Detail	Basic Bus Weight Change, pounds: Total, This Item	Payload Weight Change, pounds	Effect on Basic Bus Performance	Remarks	Study Configurations Incorporating Change
1	Remove these items from basic bus	High gain planar array antenna	8.90			Restricts nominal information bandwidth available for data transmission to earth to about 1000 cps. Restricts lunar survival to a few hours, depending on sun angle at touchdown.	Spacecraft survival could be as much as several earth days depending on time of landing and battery conditions at touchdown. A potential increase in reliability and simplification of operational control of spacecraft attitude is expected.	1B 2B 3B 4B 4B/90
		Movable solar panel	8.50					
		Antenna/solar panel positioner	23.63					
		Supporting mast and structure	4.53					
		Cables and connectors for above items	5.34	-50.90				
	Add these items to basic bus	Fixed 10-square foot solar panel	9.80					
		Supporting structure and cables for panel	3.00	+12.80				
				-38.10	+38.10			
2	Remove the following items:	Approach TV camera 4	6.80			No capability to take approach TV pictures.	Not normally required for mission support.	1A through 4D
		Television auxiliary	1.64					
		Cables and supporting brackets	2.65	-11.09	+11.09			
3	Repackage central signal processor, engineering signal processor, signal processor auxiliary and low data rate auxiliary into one unit.			-1.5	+1.5	None	Combines the circuit functions of several units into one.	1A through 4D
4	Remove present large battery. Replace with two smaller batteries (1400 watt-hours each) and slightly modify power circuitry.		-46 +45	-1.0	+1.0	Improved reliability; loss of a nominal 700 watt-hours of energy storage capacity.	No effect on transit and landing for 66-hour mission assuming no extended survival requirements.	1A 2A 3A 4A
5	Add one additional large battery and slightly modify power circuitry.			+46	-46	Adds 3500 watt-hours of energy storage capability.	Required for 90-hour trajectory and additional power for lunar night survival.	1D 1D/90 2D 3D 4D
6	Remove present large battery. Replace with two small (500 watt-hour) batteries. Add RTG. Add mounting structure, shielding and power control circuitry.		-46 +20 +30 +8	+12	-12	Provides adequate 90-day survival capability	Improved probability of 30- to 90-day survival will result.	1C 2C 3C 4C

together with a titanium case represents performance improvement attainable through advanced propellants and materials. Within these two main categories, the configurations are further broken down to include two categories of vernier engines, those with a limited (3.5:1) thrust ratio and those with an extended (9:1) thrust ratio. *

The two main retro engine categories, when combined with the two vernier engine subcategories, comprise a matrix of four basic propulsion combinations. Each of the four propulsion combinations is further subdivided into four different spacecraft configurations intended to illustrate the effects of changes in nonpropulsive elements on payload and injected weight. With the exception of A-21A, each of these four basic designs is shown for comparative purposes in each of the four propulsion categories. In addition, two 90-hour trajectories are included. One of these appears in the propulsion category most nearly resembling the current A-21A design. The other one appears in the propulsion category representing the greatest improvement in total payload weight capacity. In this manner the effects of a 90-hour trajectory on payload and injected weight relationships are illustrated for both ends of the spectrum of propulsion design combinations.

Including the A-21A configuration, Figure 7-9 illustrates the total injected weight and resulting maximum payload capability for 19 different spacecraft configurations. The four different basic configurations composed of various combinations of nonpropulsive spacecraft elements are further subdivided according to mission performance into either limited survival or 30/90 day survival categories. Limited survival in the context of this study is defined as survival for a period ranging from 2 hours to several earth days depending on landing location and relative sun angle at touchdown. Thus, all of the A and B configurations are intended to provide soft lunar landing and a relatively short period of postlanding assessment of spacecraft and payload condition. Spacecraft operation beyond this period is possible, but should not be expected on a routine basis. The probability of survival beyond the

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*It is also possible to examine combinations of aluminum propellant with a titanium case, beryllium propellant with a steel case, as well as each main retro combination with or without a 5.4-inch nozzle extension. These additional combinations when combined with the two vernier engine combinations and with combinations resulting from the various nonpropulsive modifications of basic bus design would produce a relatively large number of configurations many of which would be of little practical interest. Presentation of results of this magnitude have been avoided in this report because they would be unwieldy, and would also tend to obscure the principal study results in a mass of quantitative detail.

immediate postlanding period is much higher for the A configurations than for the B configurations because of the absence in the latter design of a movable solar panel which can track the sun. Neither of these designs, however, is intended to provide optimum lunar day thermal operation or survival beyond the day/night terminator.

The C and D configurations will provide for 30- to 90-day survival of the spacecraft. No design change is implied in providing 90-day as compared with 30-day survival. The only difference is a lower probability of survival for a 90-day case. The C configuration incorporating an RTG is most attractive for this type of mission because it permits a higher payload weight to be realized for a given injected weight, and its probability of 90-day survival is higher.

The previously discussed Table 7-1 illustrated the tradeoffs between basic bus and payload weight for the various design modifications considered. Table 7-2 summarizes each configuration studied and presents a listing of important weight characteristics. The definitions of the various weight figures employed are significant in this table. Injected weight is defined as total weight of the spacecraft immediately after separation from the Centaur launch vehicle including expendables, basic bus, and payload. Expendable weight is defined as the total weight of all items which are or could be expended during the course of transit and landing. This includes total weight allocations of nitrogen, helium, vernier engine oxidizer, vernier engine fuel, the AMR, main retro engine propellant, and the main retro engine case together with associated wiring, insulation, and heaters. The definition of expendable items, as employed in this report, arbitrarily includes all of the expendable items loaded irrespective of their actual use prior to touchdown. For example, the unusable vernier engine fuel and oxidizer is included in the above category. Helium, which is not actually expended in terms of weight, is included for purposes of definition because it is more appropriate in this category rather than as an item of basic bus equipment. Dry landed weight is defined as injected weight minus expendable weight. It is also the sum of basic bus weight plus payload weight. Basic bus weight is defined as the total touchdown weight of the spacecraft (not including unused expendables) minus the weight of the payload. Table 7-3 shows a detailed weight breakdown of the configurations presented in Figure 7-9 . It should be noted in all of the weight presentations including A-21A that the weight allocation to basic bus contingency has, for study purposes, been assigned to the payload. This has been done to make direct comparisons with A-21A more meaningful since none of the study configurations include a contingency weight allocation.

Because the mechanical requirements and configurations of the payload have not been defined for this study, it is impractical to arrange the elements of the basic bus to provide center of gravity compensation. Accordingly, for structural and center of gravity considerations it has been

TABLE 7-2. DETAILED SUMMARY OF CONFIGURATIONS STUDIED

		PROPULSION DESIGN PARAMETERS								ELECTRICAL POWER SYSTEM					TELECOM-MUNICATIONS		ALTITUDE MARKING RADAR		SPACECRAFT WEIGHT, Pounds						
		Main-Retro Rocket Engine						Vernier Engine Type		Batteries		RTG + Two 10-pound Batteries	Solar Panel		High Gain Antenna				Injected Weight =	Weight of Expendables +	Dry Landed Weight				
		Propellant		Nozzle Extension		Thermal Control															Dry Landed Weight				
Configuration	Trajectory Time, hours	Aluminum with Steel Case	Beryllium with Titanium Case	0 inch	5.4 inches	Same as A-21A	Modified (Additional Insulation and Heaters)	Extended Throttling Range	Restricted Throttling Range	Two Large (93 pounds)	Two Small (45 pounds)		Fixed	Movable	Included	Not Included	A-21A Dish Antenna	Modified Antenna (Planar Array)	Injected Weight	Weight of Expendables	Basic Bus	Payload	Dry Landed Weight Totals	Remarks	pl. Wing
A-21A	66	X		X		X			X	See Remarks				X	X		X		2150	1549	527	74	601	One 46-pound battery on A-21A	.0344
1-A	66	X		X		X			X		X			X	X		X		2526	1800	533	193	726	Limited survival	.0767
1-B	66	X			X	X			X		X		X			X		X	2568	1829	496	243	739	Limited survival, narrow band telemetry	.0945
1-C	66	X			X	X			X			X		X	X			X	2568	1829	545	194	739		.0754
1-D	66	X			X	X			X	X				X	X			X	2568	1829	582	157	739		.0611
1-D/90	90	X			X	X			X	X				X	X			X	2628	1843	591	194	785	90-hour trajectory	.0737
2-A	66	X		X		X		X			X			X	X		X		2563	1811	553	199	752	Limited survival	.0775
2-B	66	X			X	X		X			X		X			X		X	2604	1839	517	248	765	Limited survival, narrow band telemetry	.0940
2-C	66	X			X	X		X				X		X	X			X	2604	1839	566	199	765		.0764
2-D	66	X			X	X		X		X				X	X			X	2604	1839	603	162	765		.0673
3-A	66		X	X			X		X		X			X	X		X		2486	1739	536	211	747	Limited survival	.0848
3-B	66		X		X		X		X		X		X			X		X	2527	1766	500	261	761	Limited survival, narrow band telemetry	.103
3-C	66		X		X		X		X			X		X	X			X	2527	1766	549	212	761		.0837
3-D	66		X		X		X		X	X				X	X			X	2527	1766	586	175	761		.0693
4-A	66		X	X			X	X			X			X	X		X		2531	1753	557	221	778	Limited survival	.0872
4-B	66		X		X		X	X			X		X			X		X	2572	1779	522	271	793	Limited survival, narrow band telemetry	.103
4-B/90	90		X		X		X	X			X		X			X		X	2639	1789	532	318	850	90-hour trajectory	.1201
4-C	66		X		X		X	X				X		X	X			X	2572	1779	571	222	793		.0864
4-D	66		X		X		X	X		X				X	X			X	2572	1779	608	185	793		.0712

assumed for study purposes that the payload weight is appropriately distributed about the spaceframe. This is consistent with the prime objectives of the study which are to examine limits of practical feasibility rather than to define a specific spacecraft configuration for a given payload.

PAYLOAD WEIGHT AS FUNCTION OF INJECTED WEIGHT

The same 18 configurations (excluding A-21A) presented in Figure 7-9 are also illustrated in terms of the effects of variations in injected weight on payload weight. These results are shown in Figure 7-10. This parametric presentation will be useful in considering tradeoffs and effects resulting from potential variations in the injected weight capabilities of the Atlas/Centaur launch vehicle.

The payload weights presented were obtained by subtracting the following items from the dry landed weight data (presented in Figures 3-10 and 3-11):

1) Weight changes in the spaceframe structure, landing gear, and crushable blocks (which vary with dry landed weight) as discussed in Section 5.

2) Weight changes in the vernier propellant tanks which vary with vernier propellant usage.

3) Total weight of basic bus items, other than those in items 1 and 2 above, which are appropriate for each configuration.

The performance/weight tradeoffs illustrated in Table 7-1 are also valid for these parametric cases. The detailed configuration definitions and weight breakdown of Tables 7-2 and 7-3 are valid except for weights allocated to items 1 and 2 above.

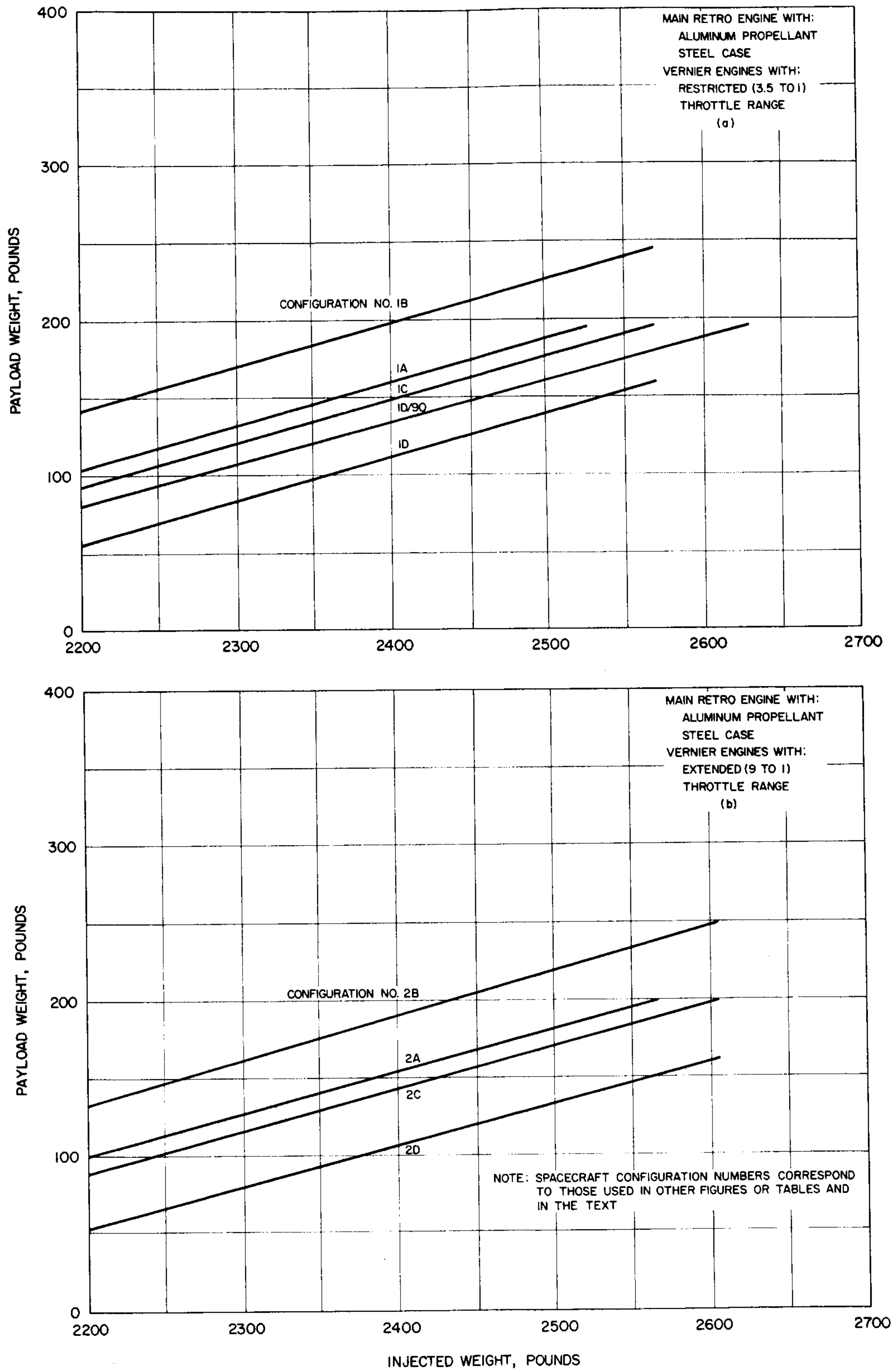


Figure 7-10. Parametric Curves - Payload Weight versus Injected Weight

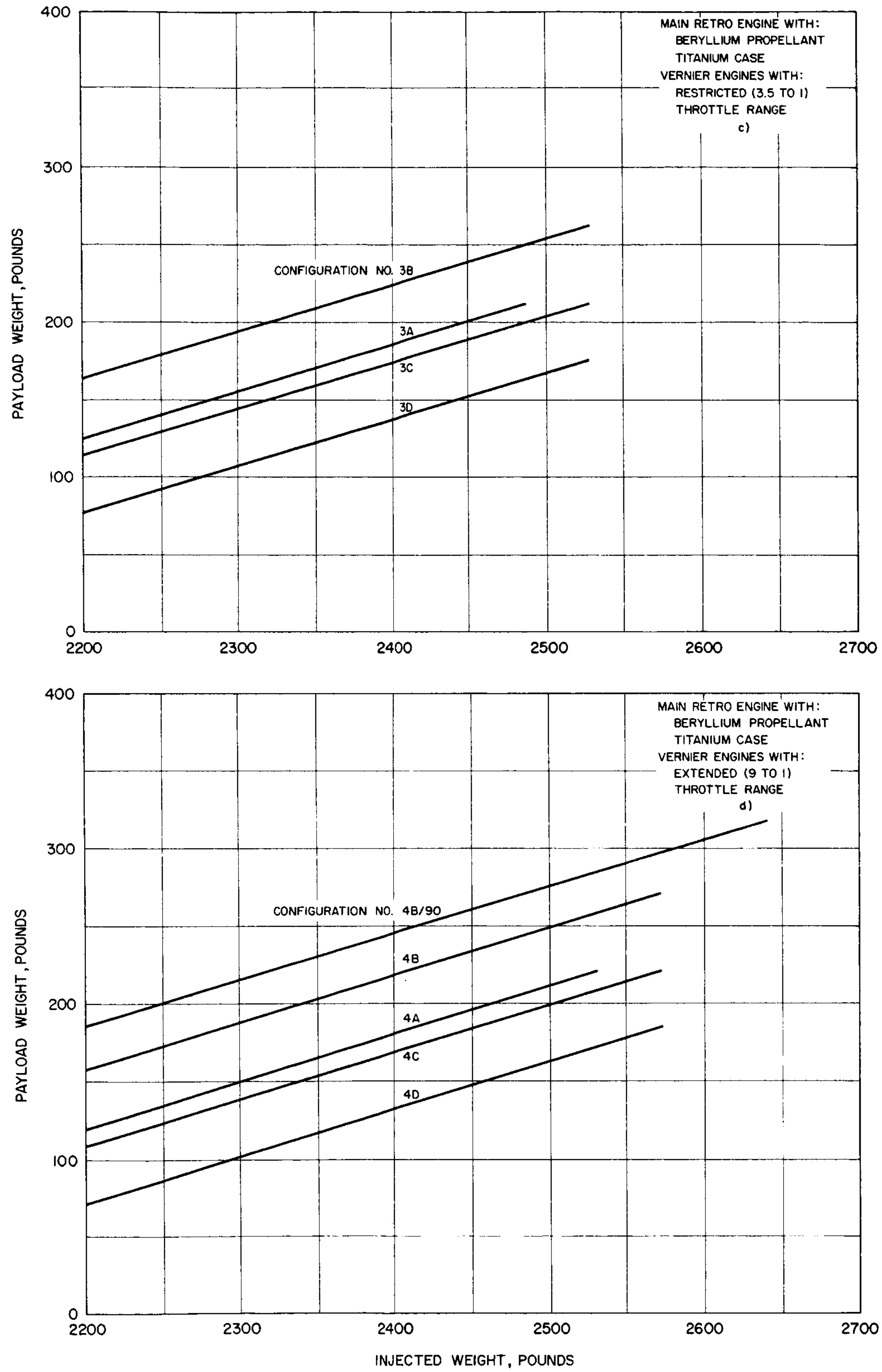


Figure 7-10. Parametric Curves - Payload Weight versus Injected Weight

TABLE 7-3. DETAILED WEIGHT BREAKDOWN OF CONFIGURATIONS STUDIED

Configuration Number	Injected Weight, pounds	Weight of Expendables, pounds						Total Weight of Expendables, pounds
		Main-Retro	Helium	Nitrogen	Propellant		AMR	
					Retro	Vernier		
A-21A	2150	142.9	2.5	4.5	1236.1	153.6	8.9	1548.6
1-A	2526	151.9	2.5	4.5	1446	186.6	8.9	1800.4
1-B	2568	152.8	2.6	4.5	1470	189.6	9.5	1829.0
1-C	2568	152.8	2.6	4.5	1470	189.6	9.5	1829.0
1-D	2568	152.8	2.6	4.5	1470	189.6	9.5	1829.0
1-D/90	2628	153.6	2.8	4.5	1470	202.2	9.5	1842.6
2-A	2563	152.0	2.7	4.5	1446	196.7	8.9	1810.8
2-B	2604	153.0	2.7	4.5	1470	199.1	9.5	1838.8
2-C	2604	153.0	2.7	4.5	1470	199.1	9.5	1838.8
2-D	2604	153.0	2.7	4.5	1470	199.1	9.5	1838.8
3-A	2486	157.9	2.6	4.5	1377	188.2	8.9	1739.1
3-B	2527	158.6	2.6	4.5	1400	191.3	9.5	1766.5
3-C	2527	158.6	2.6	4.5	1400	191.3	9.5	1766.5
3-D	2527	158.6	2.6	4.5	1400	191.3	9.5	1766.5
4-A	2531	157.9	2.8	4.5	1377	202.2	8.9	1753.3
4-B	2572	157.9	2.8	4.5	1400	204.7	9.5	1779.4
4-B/90	2639	157.9	2.9	4.5	1400	214.5	9.5	1789.3
4-C	2572	157.9	2.8	4.5	1400	204.7	9.5	1779.4
4-D	2572	157.9	2.8	4.5	1400	204.7	9.5	1779.4

TABLE 7-3 (continued)

Basic Bus, pounds						Dry Landed Weight, pounds		
Flight Control	Electronics	Electrical Power	Mechanisms	Spacecraft Vehicle	Vernier Propulsion	Total Basic Bus	Payload	Total Dry Landed Weight, pounds
48.2	103.1	54.9	27.8	218.4	74.8	527.2	74.2*	601.4
48.2	94.8	53.5	27.8	230.8	77.2	532.3	193.3	725.6
48.2	85.9	54.8	4.1	225.5	77.7	496.2	242.8	739.0
48.2	94.8	64.5	27.8	232.2	77.7	545.2	193.8	739.0
48.2	94.8	101.5	27.8	232.6	77.7	582.4	156.6	739.0
48.2	94.8	101.5	27.8	239.2	80.1	591.6	193.8	785.4
48.2	94.8	53.5	27.8	238.5	90.1	552.9	199.3	752.2
48.2	85.9	54.8	4.1	233.1	90.7	516.8	248.4	765.2
48.2	94.8	64.5	27.8	239.8	90.7	565.8	199.4	765.2
48.2	94.8	101.5	27.8	240.2	90.7	603.0	162.2	765.2
48.2	94.8	53.5	27.8	234.0	77.5	535.8	211.1	746.9
48.2	85.9	54.8	4.1	228.6	78.1	499.7	260.8	760.5
48.2	94.8	64.5	27.8	235.3	78.1	548.7	211.8	760.5
48.2	94.8	101.5	27.8	235.5	78.1	585.9	174.6	760.5
48.2	94.8	53.5	27.8	241.6	91.3	557.2	220.5	777.7
48.2	85.9	54.8	4.1	236.7	91.7	521.4	271.2	792.6
48.2	85.9	54.8	4.1	244.9	93.5	531.4	318.3	849.7
48.2	94.8	64.5	27.8	243.4	91.7	570.4	222.2	792.6
48.2	94.8	101.5	27.8	243.7	91.7	607.7	184.9	792.6

* Includes basic bus contingency weight allocation

8. RELIABILITY

LUNAR SURFACE SURVIVAL

The probabilities of lunar survival of the Block II basic bus have been calculated. The calculations are based on A-21A reliability estimates, and assume the transmitter duty cycle to be the same as for A-21A. Reliability estimates for the power subsystem are based on the RTG-solar panel-battery system described in Section 4 of this report. The probabilities of survival for 30 days, 90 days, and 2 years, assuming a fully operative spacecraft on landing, are shown in Table 8-1.

TABLE 8-1. PROBABILITIES OF SURVIVAL

Item	30 Days	90 Days	2 Years
Central command decoder	0.992	0.99	0.920
Structures	0.96	0.96	0.96
Thermal controls	0.994	0.982	0.85
Power subsystem	0.96	0.91	0.6**
Telecommunications*	0.989	0.91	0.095
Data processing	0.945	0.87	0.792
Probability of system survival	0.856	0.672	0.035**

*Corrected for standby redundancy of the transmitters; previous estimates were based on hard redundancy.

**Assumes perfect power sources.

The calculation of the probability of 2-year survival assumed no failures in the power sources (RTG, solar panel, batteries) because these elements would probably be different from their counterparts in the shorter missions and data was not available. Even with this unrealistic assumption, an unacceptably low (0.035) probability of survival is estimated. The major source of unreliability is seen to be the telecommunications subsystem. No practical amount of redundancy will provide an acceptably high probability of 2-year survival. The 2-year mission is not considered practical without major spacecraft redesign.

LANDING AND NO SURVIVAL

Predicted probabilities of successful flight and landing are shown in Table 8-2.

TABLE 8-2. PREDICTED PROBABILITIES OF SUCCESSFUL FLIGHT AND LANDING

Item	One Midcourse Correction		Two Midcourse Corrections	
	66-Hour Transit	90-Hour Transit	66-Hour Transit	90-Hour Transit
Single battery	0.808	0.783	0.802	0.777
Two batteries (recommended in Section 4 of this report)	0.836	0.813	0.831	0.808

The two-battery system has the additional advantage of being able to provide a significant period of lunar surface operation if neither battery fails prior to touchdown.

REDUNDANCY CONSIDERATIONS

The several spacecraft subsystems were examined to determine whether it would be appropriate to increase or decrease the amount of redundancy that is employed, relative to the degree of redundancy provided in A-21A. Table 8-3 summarizes subsystem reliabilities through transit and touchdown with and without redundancy. The subsystems are listed in

TABLE 8-3. RELIABILITY OF SURVEYOR BY FUNCTIONAL BLOCKS SHOWN FOR REDUNDANT AND NONREDUNDANT SYSTEMS

66 Hours — Two Midcourse Maneuvers

Functional Block	Nonredundant	Redundant Where Practical
Telecommunications and central command decoder	0.936	0.99 (presently exists)
Flight controls	0.937	—
Propulsion	0.95	—
Data processing	0.96	0.995 (presently exists)
Electrical power	0.962	0.996 (suggested)
Mechanisms	0.969	—
Structures	0.977	—
Thermal controls	0.998	—

ascending order of reliability; thus redundancy is most desirable for subsystems at the top of the listing.

Continued use of redundancy in telecommunications and data processing is recommended. Telecommunications is in series with all other spacecraft operations. It is important that communications be maintained for failure diagnosis even if other subsystems fail, and significant reliability improvement is provided for relatively small weight.

Flight control has the second lowest reliability. Unfortunately, this is the most complex subsystem, and adding redundancy would require the addition of complex logic and switching circuitry. Further, since the subunits do not necessarily fail catastrophically but fail because of gradual degradation, it may be necessary to provide triple redundancy so that a failure can be detected by comparison techniques. Such redundancy is impractical.

Weight limitations make propulsion redundancy impractical.

Electrical power reliability can be significantly improved by the use of two batteries, and is recommended for Block II.

9. LANDING ACCURACY

The accuracy with which the spacecraft can be landed at a specified site depends on the following three factors: the accuracy with which the spacecraft orbit can be determined from radio tracking, the accuracy of the in-flight maneuver(s) intended to correct injection errors, and the incidence angle of the approach asymptote at the moon. Orbit determination errors and maneuver execution errors are measured by their effect on the impact parameter or B-vector of the transit trajectory. B-plane errors are translated into landing location errors by taking into account the differential focusing effect of the moon, a factor which depends on the incidence angle. Except where it is stated otherwise, all results apply to 66-hour trajectories.

It is assumed that the midcourse maneuver(s) are to be performed while the spacecraft is in view of the Goldstone tracking station and that the maneuver is to be based on an orbit computed with the aid of at least 1 hour of Goldstone tracking during the same view period. It is also assumed that the statistics governing the execution errors are identical to those for the A-21A spacecraft.

The orbit-determination process, however, will differ in several respects. In addition to Goldstone, there will be DSIF stations at Johannesburg, Canberra, and Madrid. It is not anticipated that the latter station will be operational for the A-21A spacecraft. In addition, all stations will be equipped with atomic frequency standards, thus providing them with a doppler accuracy capability now possessed only by Goldstone. Finally, the possibility that the spacecraft will carry a transponder to provide turn-around ranging capability will be considered.

The estimates of orbit determination accuracy are based on the assumption that independent doppler, hour-angle, and declination measurements are taken every 60 seconds. The 1σ doppler accuracy is taken as 0.05 cps. Angular accuracy is taken as 0.14 degree. (A value of 0.14 degree is used despite the single measurement angular accuracy of 0.01 degree, since the correlation time for these errors which are due primarily to antenna deflections is assumed to be 300 minutes.) The error in the ranging system is essentially a bias, which would be constant throughout the flight and not greater than 15 meters in magnitude. Since the JPL Mariner Orbit Determination Program, which was used to obtain the orbital accuracy estimates, is not equipped to solve for biases at present, it was not possible to evaluate properly the improvement due to the inclusion of range data. Instead, it was assumed that independent range measurements, with a 1σ accuracy of 15 meters, are taken every 60 seconds. Increasing the sampling interval of the range measurements to 10 minutes degraded the orbital accuracy only very slightly. It may therefore be concluded that the quoted improvements due to ranging are not strongly dependent on the assumed frequency of the range measurements.

It has been found empirically that the 99-percent point of the miss distribution resulting from execution errors alone, when averaged over the distribution of required first maneuvers, is approximately equal to the 99-percent point of the miss distribution for a fixed maneuver of 1σ magnitude. The execution errors have therefore been evaluated for a maneuver having a critical-plane component of 10 meters per second, the figure of merit for the Centaur injection guidance system. It is further assumed that the maneuver component in the noncritical direction is considerably smaller than 10 meters per second, so that the resultant maneuver may be taken to be 10 meters per second. For a maneuver of this magnitude, the execution error is nearly spherically distributed, each component having a 1σ value of 0.133 meter per second. The B-plane sensitivity to execution errors at a typical first maneuver time of 16 hours after injection is approximately 200 kilometers/meters/second. Therefore each B-plane error component has a 1σ value of 27 kilometers for execution errors alone.

The contribution of orbit-determination errors to the B-vector uncertainty is a function of the launch azimuth and the lunar declination at encounter. The semimajor axis of the B-plane error ellipse (1σ) was found to vary from 8 to 16 kilometers over a range of trajectories when only doppler and angular data were used. When range data was added, there was considerably less variation, the semimajor axis being on the order of 4 kilometers. In either case, however, the execution errors are dominant, with the semimajor axis of the resultant B-plane error ellipse lying between 28.2 and 31.4 kilometers. For simplicity, in the ensuing discussion the B-plane error resulting from the first maneuver will be assumed to be circularly distributed, the 1σ value of each component being 30 kilometers.

This error must be corrected by the second maneuver, which is typically made 40 hours after injection. At this time, the B-vector sensitivity has been reduced to 100 kilometers/meter/second, so that the required (critical-plane) maneuver has a magnitude of 0.3 meter per second. The execution error for a maneuver of this small magnitude is primarily a bias or shut-down error in the direction of the maneuver, having a 1σ value of 0.03 meter per second. The corresponding B-plane error is 3 kilometers.

The semimajor axis of the B-plane error ellipse (1σ) due to orbit determination errors varies from 2.5 to 8.5 kilometers without range data, but was found to be no greater than 1.7 kilometers, and often considerably smaller, when range data was included.

The semimajor axis of the resultant B-plane error ellipse in the absence of range data can therefore be as large as 9 kilometers or as small as 3.9 kilometers. With range data, it appears that the 1σ accuracy may be as small as 3 kilometers. Thus the inclusion of range data offers a potential improvement in landing accuracy by a factor varying from 1.3 to 3, depending primarily on the trajectory.

B-plane errors are translated into landing errors by applying a multiplication factor which varies from 0.61 at an incidence angle of 0 degree to 0.91 at 45 degrees. Upper and lower bounds of the image, on the surface of the moon, of the semimajor axis of the B-plane error ellipse (1σ) are given in Figure 9-1 for the case of no range data. The accuracy when range data is included is also given in Figure 9-1.

The landing accuracy afforded by a 90-hour trajectory depends on the time of the second maneuver. If it is performed during the second Goldstone pass, the orbit-determination accuracy, expressed in terms of the injection errors, should be relatively unchanged from the 66-hour case. In terms of the B-vector, however, the errors will be greater by approximately a factor of 90/66 = 1.36. The effect of execution errors will be twice as great as those for the 66-hour case. The net result is a B-plane semimajor axis which varies between 6 and 13 kilometers in the absence of range data, instead of between 4 and 9 kilometers. With range data, the increase is from 3 to 6 kilometers.

If the maneuver in the 90-hour case should be performed during the third Goldstone pass, the execution errors will have approximately the same effect as in the 66-hour case. However, because of the increased tracking time, it should be possible to reduce the orbit-determination errors somewhat, so that a maximum total B-plane semimajor axis of perhaps 5 kilometers would result.

It is evident from the foregoing discussion that the benefit to be derived from the use of range data depends on its availability up to the second maneuver time. Table 9-1 shows that, at the lunar distance, the ground-to-spacecraft link provides a signal-to-noise ratio which is 16.8 db less than that required for the ranging code. The communication range corresponding to this negative margin is only 59,000 kilometers. The required increase in the signal-to-noise ratio could be obtained by utilizing the planar array which has a gain of 24 db at the up-link frequency. This would require the addition of RF switching circuitry to the spacecraft, and would in general necessitate a spacecraft attitude change whenever a range measurement is to be made so that the antenna beam intersects the earth.

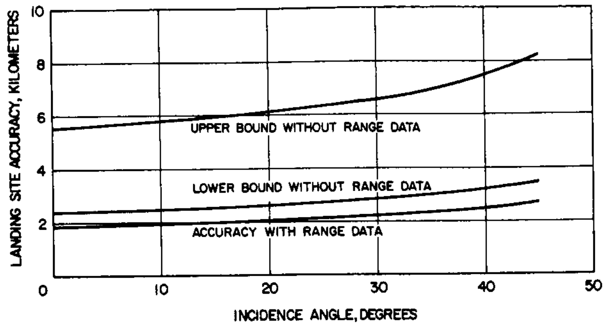


Figure 9-1. Landing Site Accuracy for 66-Hour Trajectory after Second Maneuver

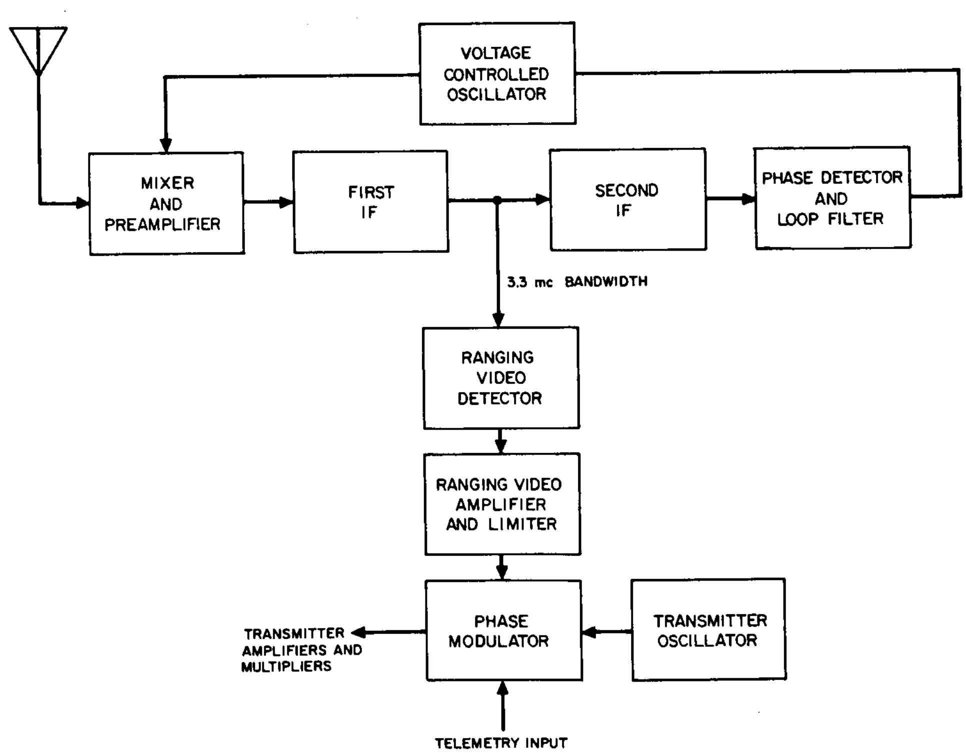


Figure 9-2. Block Diagram for Block II Spacecraft Transponder with Turnaround Capability

TABLE 9-1. TRANSMISSION OF RANGING CODE FROM THE GROUND STATION TO SPACECRAFT

Transmitting System (DSIF)	
Carrier	
Transmitter power	70 dbm (10 kilowatts)
Circuit loss	0.4 db
Antenna gain, 210-foot diameter*	58 db
Modulation loss	6.8 db
Propagation loss (407,000 kilometers)	211.15 db
Receiving System (Spacecraft)	
Antenna gain	0 db
Circuit loss	3.3 db
Noise spectral density	-161.0 dbm/cps
Noise bandwidth	470 cps
Required signal-to-noise ratio	7 db
Available signal-to-noise ratio	40.64 db
Sum of negative tolerances	15.31 db
Minimum margin = available — required — negative tolerance	18.33 db
Ranging Code	
Modulation loss	1.9 db
Noise bandwidth	3.3 mc
Required signal-to-noise ratio	10.5 db
Available signal-to-noise ratio	6.34 db
Sum of negative tolerances	12.66 db
Minimum margin = available — required — negative tolerance	-16.82 db

*Scheduled for operation in 1967.

The design parameters for the spacecraft-to-ground link are shown in Table 9-2. It is assumed that the exponential signal-to-noise improvement resulting from passing the ranging code through the limiter shown in the block diagram of Figure 9-2 is ample for modulation of the transmitter. With a carrier modulation index of 1.45 radians (peak), there is sufficient sideband power available at the DSIF for acquisition of the ranging code. However, this high modulation index precludes simultaneous telemetry. Since only 10 seconds are required to make a range measurement, this should present no difficulty.

To provide a ranging capability, the following changes and additions must be made in the spacecraft transponder:

1) An increase in the bandwidth of the first IF amplifier in the receiver to 3.3 mc.

2) The addition of

   An isolation amplifier

   A balanced detector

   Video amplifiers and a limiter

The estimated transponder weight increase is 1/2 pound.

Since the improved landing accuracy provided by ranging does not approach the order of accuracy that is desired for many Block II missions, it is questionable whether the spacecraft changes required by ranging are justified.

TABLE 9-2. TRANSMISSION OF THE RANGING CODE FROM THE SPACECRAFT TO THE DSIF

Transmitting System (Spacecraft)	
Carrier	
Transmitter power	20 dbm (100 milliwatts)
Transmission line losses	3.5 db
Antenna gain	0 db
Modulation loss (1.45 radians peak)	5.3 db
Space propagation loss (407,000 km)	211.86 db
Receiving System (DSIF)	
Antenna gain (210-foot diameter*)	60 db
Receiver noise density	-176.42 dbm/cps
Loop noise bandwidth ($2B_{L0}$ = 12 cps)	10.8 db
Required signal-to-noise ratio	6 db
Available signal-to-noise ratio	24.78 db
Sum of negative tolerances	11.89 db
Minimum margin = available − required − negative tolerance	6.89 db
Code	
Modulation loss	2.2 db
Required signal-to-noise ratio (in $2B_{L0}$ = 5 cps)	20 db
Available signal-to-noise ratio	31.71 db
Sum of negative tolerances	11.69 db
Minimum margin = available − required − negative tolerance	0.02 db

*Scheduled for operation in 1967.

10. EXTENSION OF SURVEYOR LANDING AREA CAPABILITY

INTRODUCTION

There are a number of concepts for extending the landing area capability of Surveyor, ranging from fairly simple to drastically different modifications. While the more complicated approaches can potentially give a greater area coverage than the basic Surveyor concept of deboosting and landing directly from an impacting trajectory, they also require a good deal greater ($\sim$1000 fps) characteristic velocity and have other propulsion and guidance requirements not compatible with the Surveyor design. On the other hand, nearly horizontal incidence angle capability is required of the present Surveyor if it is to achieve a substantially greater area capability. For example, to cover the entire area of ±10 degrees in latitude and ±60 degrees in longitude, the incidence angle capability required is between 0 (normal approach) and approximately 75 degrees. In theory, such increased incidence angle capability does not involve additional fuel cost but this is not entirely so in practice. Certain changes must be made in the spacecraft mechanization, which in turn causes some increases in both main retro and vernier fuel requirements. Additionally, the landing accuracy of higher incidence angles is degraded because of increased sensitivities to errors in the impact parameter. There is also the necessity of biasing the main-retro thrusting direction from the approach velocity by some small angle in order to bring the nominal burnout velocity vector into acceptable bounds.

Calculations are made for a separated weight of 2150 pounds rather than the heavier weights to show the fuel costs above the present A-21A design. Propellant requirements can be expected to increase in proportion to the separated weight for the heavier vehicles with all other conditions held fixed.

MODIFICATIONS FOR PRESENT SURVEYOR

The modifications necessary to accommodate the increased incidence angles lie in 1) proper pointing of the altitude marking radar (AMR) prior to main-retro ignition, 2) proper pointing of the RADVS after main-retro burnout, and 3) equipment additions and changes necessary to accomplish the above.

Preignition AMR Pointing

The present AMR is body-fixed and its beam is directed along the thrusting direction which in turn must be very nearly colinear with the relative approach velocity vector at ignition. For approach incidence angles less than 45 degrees off vertical, the AMR has a slant range capability adequate to provide a proper marking signal. For approach incidence substantially greater than 45 degrees, not only is the range capability insufficient but the accuracy of the mark is seriously degraded because of the finite beam width. Therefore, a way must be found for obtaining a mark with the AMR pointed either vertically, or at angles less than 45 degrees, and at altitudes that will allow the main-retro phase to terminate above the vernier descent contour. The following are two simple methods:

Method A – Canting of AMR (Figure 10-1a)

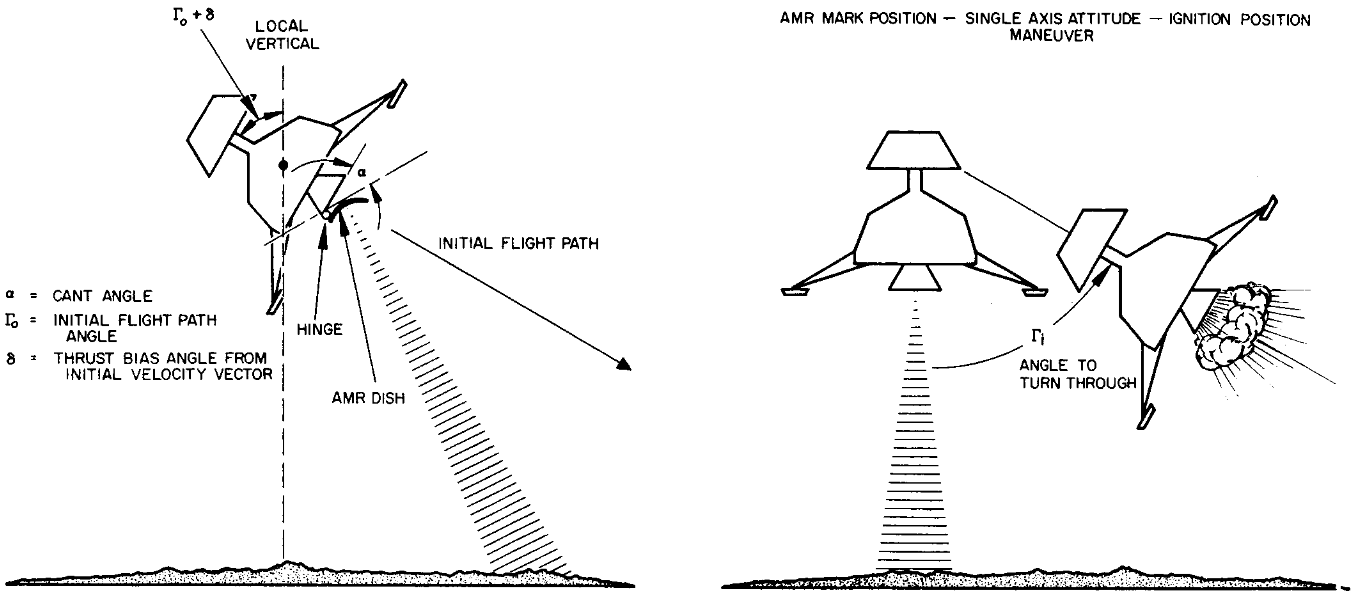
In this method, the spacecraft roll axis is commanded to the desired thrusting direction minutes before ignition, as in the present Surveyor concept. A roll maneuver is necessary to rotate the vehicle pitch axis (or yaw axis) into a position perpendicular to the approach trajectory plane. The AMR dish, modified to be rotatable about the pitch axis (or yaw axis), can be commanded to either of two hinged positions: 1) $\alpha = 0$ or 2) $\alpha =$ some fixed angle such as 45 degrees, depending on the approach angle. Thus, for large incidence angles, the second position will be commanded minutes prior to retro ignition so that the AMR beam will be directed towards the lunar surface at an acceptable look angle. Once the mark is obtained, the rest of the retro-ignition sequence remains as in the present Surveyor.

Method B – Pre-Retro Maneuver (Figure 10-1b)

In this method, the thrust axis is directed at first along the local vertical. The look angle of the AMR is therefore satisfactory for the generation of the mark. On receiving the marking signal, the verniers are turned on at approximately minimum thrust and the roll axis is turned into the desired thrusting direction by virtue of a stored pitch (or yaw) gyro command and a timing signal, both of which have been computed and sent from earth prior to the mark itself. At the completion of the attitude maneuver, the verniers are operated at mid-thrust and the main retro is ignited.

Post Burnout Attitude Erection Maneuver

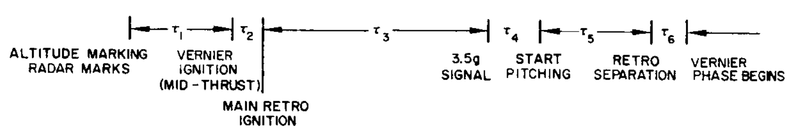
Regardless of the method chosen for pointing the AMR, the thrust axis attitude at the time of main-retro ignition is always nearly coincident with the velocity vector. Since this attitude must persist for the duration of main-retro thrusting, doppler velocity radar acquisition at burnout becomes quite questionable for large incidence angles. Therefore, an attitude erection maneuver must be made after main retro burnout, probably initiated a fixed time delay after the 3.5 g acceleration signal. It is highly desirable that this maneuver return the thrust axis to the lunar vertical. The accuracy of such an erection maneuver is unimportant, the order of 5 degrees being quite satisfactory.



a) Method A - Canting of AMR

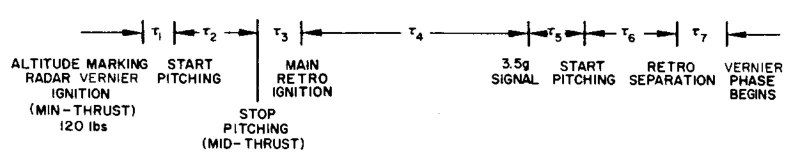
b) Method B - Pre-Retro Maneuver

Figure 10-1. Methods of Obtaining Accurate Marking Signal



τ_1 = COMMANDED DELAY
τ_2 = 1 sec
τ_3 = 40 sec
τ_4 = 3 sec
τ_5 = $\frac{(\Gamma_i)^\circ}{5}$ + 1 sec
τ_6 = 2 sec

a) Method A



τ_1 = COMMANDED DELAY
τ_2 = $\frac{\Gamma_i^\circ}{5}$ + 1 sec
τ_3 = 1 sec
τ_4 = 40 sec
τ_5 = 3 sec
τ_6 = $\frac{\Gamma_i^\circ}{5}$ + 1 sec
τ_7 = 2 sec

b) Method B

Figure 10-2. Flight Sequence

To simplify the mechanization, the rotation is to be performed about either the pitch or the yaw axis so that only one of the body-mounted integrating rate gyros need be torqued. This is why in Method A of Preignition AMR Pointing, the AMR canting is made about this same axis. In Method B, the post burnout maneuver is the exact reverse of the pre-retro maneuver and thus can use essentially the same mechanization.

During the maneuver which may last for as long as 20 seconds for a large incidence angle such as 75 degrees, the vernier fuel requirement may be held down somewhat by not requiring full thrust as in the present design. The increased thrusting time would more than compensate for the reduction in thrusting level as far as effecting a clean separation of the main-retro case is concerned.

The flight sequences for the two methods showing important events from the marking instant to the start of vernier phase are shown in Figure 10-2. Because of the necessity for the post burnout maneuver in both methods, Method B is not appreciably more complicated than Method A from the viewpoint of program storage requirement on board.

Subsystem Changes

Method A

A preliminary mechanical layout for canting of the AMR is shown in Figure 10-3. A pressure dome is inserted between the AMR assembly and the main-retro nozzle. A pin-puller and spring-loaded linkage mechanism are used for deploying the AMR assembly. When the main-retro is ignited, the entire assembly including the pressure dome is forced away by the exhaust.

The net weight increase for the deployment mechanism if made mostly with magnesium, is estimated to be 1.5 pounds. If the vacuum out-gassing of the magnesium cannot be controlled adequately, aluminum can be used with an additional weight of approximately 3/4 pound. No change to the rest of the spacecraft appears to be necessary other than the provision of an electrical connection to the deployment mechanism.

Figure 10-4 shows the AMR in the deployed position relative to the spacecraft. The spacecraft legs are well outside of the zone which must be free of objects that might distort the antenna pattern, even for a deployment angle of 45 degrees.

An extra counter of 9 or 10 binary digits is required for the post-burnout pitch (or yaw) maneuver timing at 5 deg/sec. A total of 350 parts, weighing about 3/4 pound and occupying a volume of 40 cubic inches will be needed for the counter and associated logic.

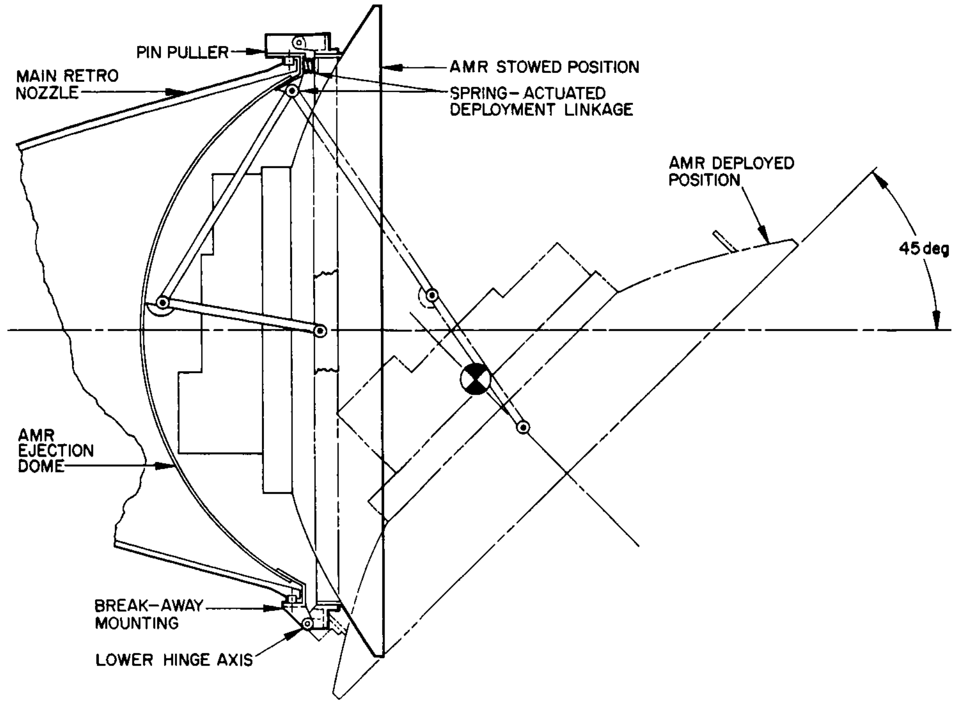


Figure 10-3. Altitude Marking Radar Deployment
Block II study

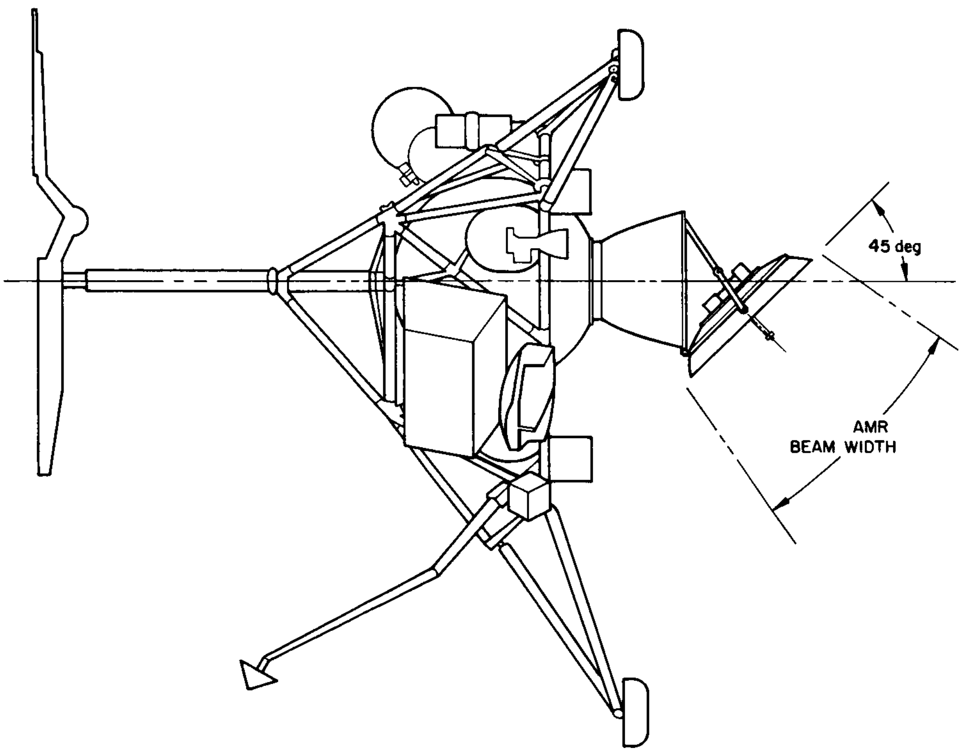


Figure 10-4. Altitude Marking Radar in Deployed Position

The rapid slewing rate of 5 deg/sec requires additional gyro-torquing circuitry estimated to consist of some 42 parts weighing a small fraction of a pound.

The additional electronic circuitry would be incorporated in flight control electronics.

Method B

Both the additional counter and the torquing circuitry requirements are identical to those of Method A. The main difference is the absence of the AMR deployment mechanism.

Approach TV and High Gain Antenna

In either Method A or B the rotational freedom in roll during the pre-retro coasting period (after disengagement from the celestially held position) is eliminated by the requirement of a single-axis post-burnout maneuver. Thus, to obtain approach TV if that should be a requirement, the antenna mast itself must be commanded to rotate about the roll axis prior to picture-taking. There will probably be the need for implementing the capability of locking the antenna in any roll position because the time required to rotate the antenna back into the present single locking position may be excessively long from the standpoint of gyro drift.

RELIABILITY COMPARISON

The following is a comparison of the relative reliability of the two methods for an approach angle of 75 degrees.

The following assumptions were made:

1) Time of preretro maneuver in Method B is about 20 seconds.

2) The verniers would be required to operate for 23 seconds longer in Method B.

3) The mechanical reliability of the mechanisms required to cant the AMR is 0.997 based on the following:

Pressure diaphram	0.999
Hinge	0.999

Pin puller (redundant squibs)	0.9999
Springs	0.9999
Linkages	0.999

4) The timing sequence would require 350 extra parts or 50 active element units.

5) The additional gyro torquing circuitry would contain 42 parts or 6 active element units.

A comparison of the reliability of each method is as follows:

	Method A, Canting AMR	Method B, Preretro Maneuver
Mechanical	0.997	—
Timing sequence	≈1	≈1
Vernier operation	—	0.9992
IRU operation	—	0.9999
Reliability of method	0.997	0.9991

ERROR ANALYSES AND PROPELLANT REQUIREMENTS

Thrust Attitude Errors

A brief analysis of the thrust axis pointing errors at the time of retro ignition produces the curve shown in Figure 10-5. The case of zero maneuver corresponds to Method A and that for 75 degrees corresponds to Method B. The respective three sigma pointing errors are 1.19 and 1.53 degrees respectively.

Burnout Velocities and Main Retro Sizing

The thrust attitude errors given above are used in the determination of the nominal maximum and minimum operating points in the burnout velocity vector diagrams of Figure 10-6 drawn respectively for Method A and Method B. These curves are drawn for an incidence angle of 75 degrees. Because of post burnout erection maneuver, the entire fan-shaped zone bounded by the 45-degree flight path limit and the 700-fps linear doppler limit is assumed to be permissible for the initial condition at the start of the vernier

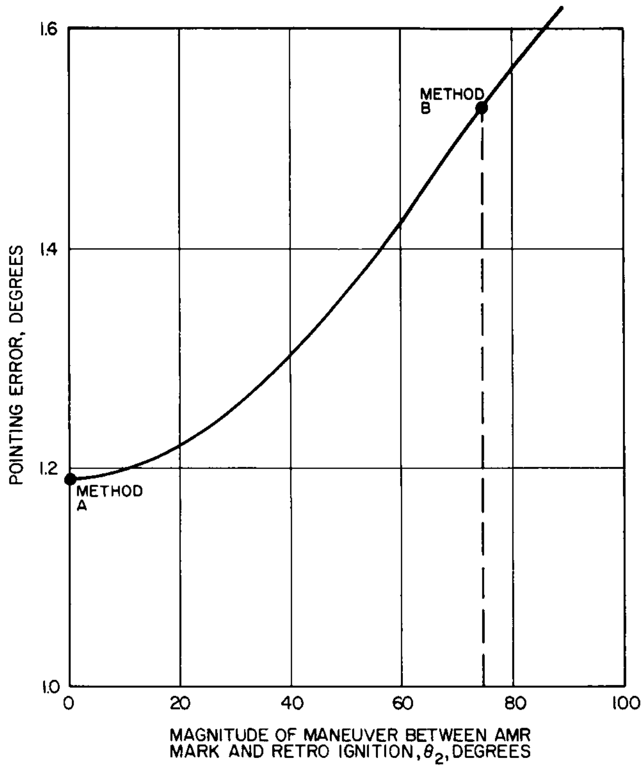
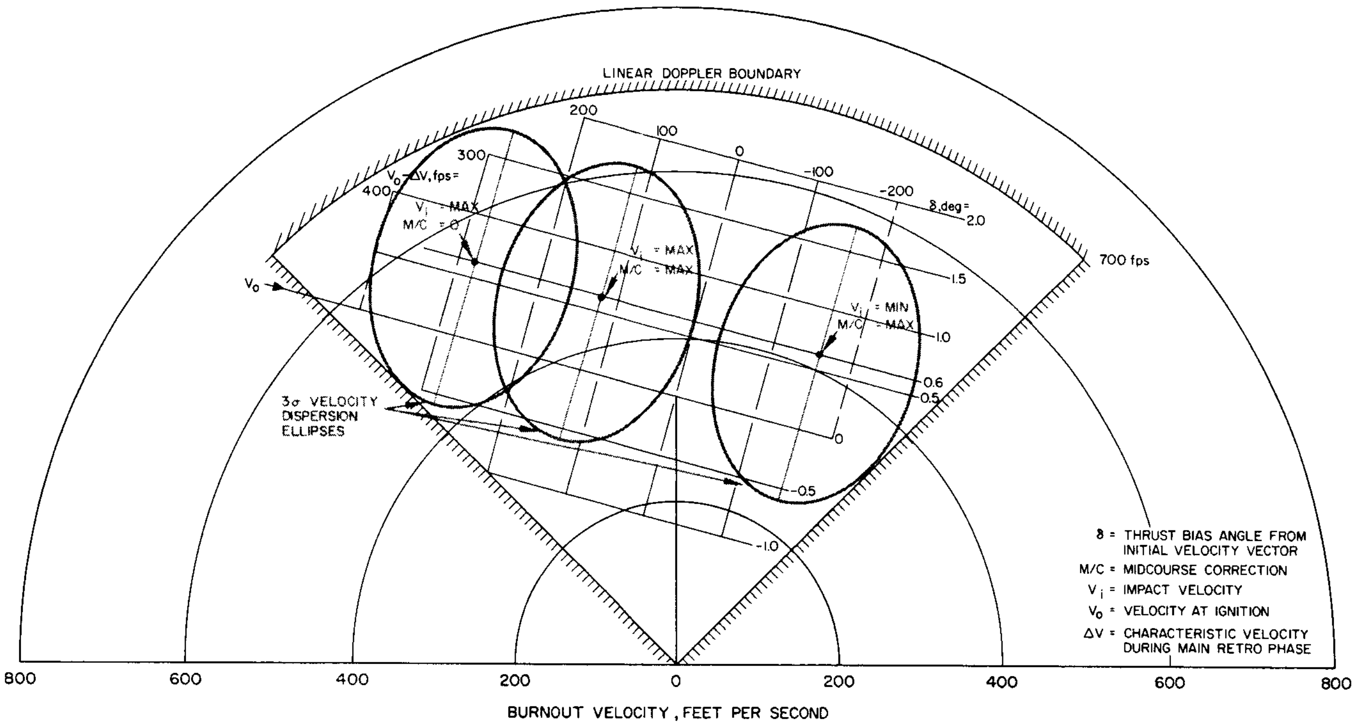
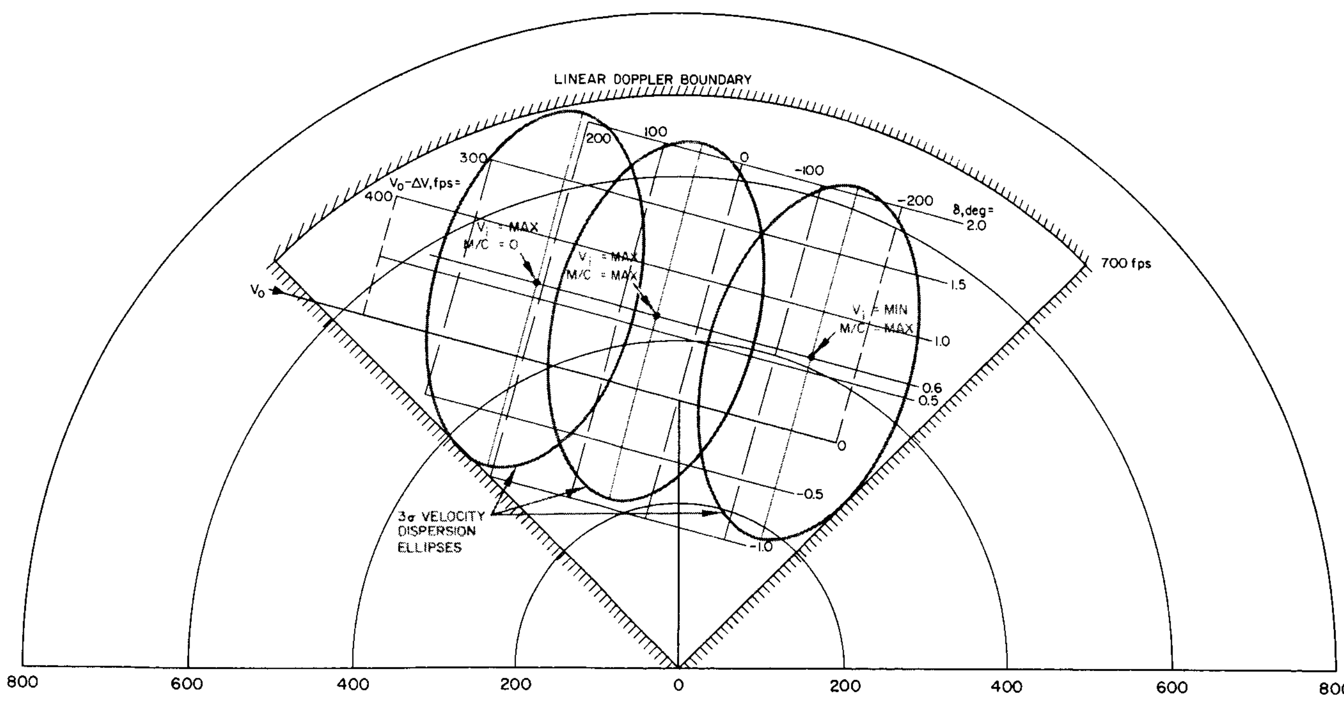


Figure 10-5. Pre-Retro Thrust Axis Pointing Error



a) Method A



b) Method B

Figure 10-6. Burnout Velocity Diagram

phase. The longitudinal-lateral velocity dispersion ellipses are for a three sigma dispersion in V_o - ΔV of 125 fps and angular errors in accordance with the discussion of thrust attitude errors. The maximum nominal values for V_o - ΔV, where V_o is the velocity at ignition and ΔV the total characteristic velocity during the main retro phase, are 285 fps for Method A and 207 fps for Method B, indicating that Method B requires a slightly heavier main-retro loading.

Based on a maximum unbraked impact speed of 2687 meters per second and a spacecraft injected weight of 2150 pounds, the following are the approximate main-retro loading requirements:

Method	Approximate Main-Retro Loading
A	1231 pounds
B	1235 pounds

Method A only shows a 4-pound advantage in main-retro loading because Method B has a slight reduction in the weight at ignition due to the vernier fuel consumption prior to retro start.

Vernier Fuel Requirements Prior to Start of Vernier Descent Phase

The following table summarizes the approximate vernier fuel requirements prior to the start of the vernier descent phase.

TABLE 10-1. VERNIER FUEL REQUIREMENTS PRIOR TO START OF VERNIER DESCENT

Phase	Method A, pounds	Method B, pounds
1. Midcourse (100 fps)	23	23
2. Preretro maneuver (120 pound thrust for 23 seconds)	0	10 (estimated)
3. Main-Retro Phase	43. 5	43. 5
Total	66. 5	76. 5

Vernier Fuel Requirements During Vernier Descent

There are several factors which will cause some increase in the vernier fuel requirement during the vernier descent phase.

1) Increase in burnout altitude dispersion due to terrain variations from the sub-AMR point to the actual landing point — applies in either Method A or Method B.

2) Increase in burnout velocity dispersion due to the closer alignment at the high incidence angles of the major axis of the burnout ellipse with the gravity loss vector — applies in either Method A or Method B.

3) Increase in burnout velocity dispersion because of degraded thrust attitude pointing accuracy — applies only in Method B.

If a 10,000-foot terrain variation is considered to be a conservative three sigma estimate, the three sigma burnout dispersion ellipse for the 75 degree approach will be approximately 27,000 feet in the altitude dimension. This represents an increase of approximately 9000 feet. The vernier fuel cost due to this cause is approximately 6 pounds.

The increase in burnout velocity because of item 2 above for Method A amounts to about 50 fps, which for a landed weight of 600 pounds amounts to about 3 pounds in vernier fuel cost. Thus, the total increase in vernier fuel requirement during the vernier descent phase is estimated to be 9 pounds for Method A. This is a worst case estimate.

For Method B, the increase in burnout velocity, with the effect of degraded pointing accuracy included is approximately 80 fps, corresponding to a net fuel increase of 5 pounds.

Total Vernier Fuel Penalties

Table 10-2 shows the approximate vernier fuel increments over the A-21A present design.

Comparison of Method A and Method B

There are clearcut propellant advantages of 4 pounds in the solid and 12 pounds in the liquid for Method A over Method B. Detracting from that is the deployment mechanism weighing 1.5 pounds. The net payload advantage of A over B is therefore 14.5 pounds.

TABLE 10-2. SUMMARY OF VERNIER FUEL INCREASES

Phase	Method A, pounds	Method B, pounds
1. Preretro maneuver	0	10
2. Main-retro phase	2.5	2.5
3. Vernier phase	9	11
Total vernier fuel increase (above A-21A requirements)	11.5	23.5

Landing Accuracy and Second Midcourse Correction for 75-Degree Approach

A serious problem with the high incidence approach is the degraded landing accuracy due to the increased sensitivity of landing site to impact parameter (commonly designated as the $\bar{B}$ vector). For a 30 meters per second 3σ correction at the nominal time of 15 hours past injection, the estimated three sigma landing dispersion is about 200 km for the 75 degree approach. If the mission objective requires a much better accuracy, a second midcourse will be necessary and an increase in the vernier fuel requirement of approximately 3 pounds must therefore be included in the budget.

11. SPACECRAFT POSTLANDING LIFTOFF AND TRANSLATION

INTRODUCTION AND SUMMARY

One portion of the Block II Surveyor Study was the consideration of providing lunar surface mobility to the entire Surveyor spacecraft. Interest in this possibility is related to the Apollo landing site certification mission, where a requirement exists to qualify a fairly extensive area (on the order of 1 kilometer or more in diameter) for LEM landing. It becomes clear that this capability is not provided by the Block I Surveyor by considering that television is the only means available to the Block I Surveyor for examinimg the nature of any portion of the lunar surface which is further removed than the reach of the surface sampler.

For a camera raised 1 meter above a perfectly smooth spherical moon, the horizon is less than 2 kilometers distant, and further objects located more than about 100 meters from the spacecraft will be badly foreshortened. Furthermore, local surface irregularities can hide from view even closely located objects. In addition, the Surveyor TV system provides a resolution of 1/4 milliradian per TV line, i.e., at 1 kilometer one TV line corresponds to 25 cm. For the site certification mission, the Surveyor TV system would at best provide only marginal performance even from an optical standpoint alone, since objects of dimensions on the order of a foot must be detected and identified over a site of more than 1 kilometer in dimension.

Thus, it was desired that the capability of the Surveyor spacecraft to lift itself from its landed position, move over some lateral distance, and land softly at a new site be assessed. The key points of this assessment were:

1) Mechanization considerations

2) Spacecraft weight (and power, if applicable) penalty, exclusive of propellant

3) Propellant requirements as a function of distance traversed

All of these points were considered, and conclusions are presented:

1) If reliability considerations and the effects of the lunar thermal environment are ignored, the Surveyor spacecraft can, with minor subsystem changes, perform the liftoff and translation mission.

2) The required hardware changes involve modifications and additions to the flight control system, Radar Altimeter and Doppler Velocity Sensor (RADVS), and vernier propellant feed system.

   However, none of the changes involved are construed as major, and the additional fixed weight involved is probably less than 10 pounds.

3) Fuel requirements are heavily dependent on spacecraft weight at liftoff, the particular mechanization scheme adopted, and distance traversed. However, a typical example of the magnitude of requirements involved is illustrated by an equivalent velocity increment requirement of 375 fps for a translated distance of approximately 600 feet, measured along the lunar surface, using the simple mechanization scheme described here. At an engine specific impulse of 285 seconds, this corresponds to 4 percent of spacecraft weight at liftoff, or 32 pounds of vernier fuel for a landed weight of 800 pounds.

4) Nevertheless, the most salient characteristic of the liftoff and translation scheme is its inherent unreliability. All of the following subsystems are required for the success of the mission.

   Vernier propulsion system including bladders, valves, thrust chambers, and lines

   Flight control system, including flight control electronics and gyros

   RADVS system

   Landing system, including gear, foot pads, shock absorbers, and crushables

These systems were not originally designed for postlanding survival, and in addition the feasibility of the postlanding liftoff and translation scheme is called into question by the following considerations:

Landing damage: Although the Surveyor spacecraft is fully qualified to land on the specified lunar surface as per JPL specification, the actual composition and structural properties of the surface are currently speculative. Hence, the probabilities of landing survival of the subsystems listed above are 1) unknown, and 2) probably quite low, if any reasonalbe degree of convervatism is to be introduced into the estimates. (It is well to remember that "survival" is meant to imply, here, a capability of functioning as per original specifications.)

Essentially all of the subsystems required to operate for the liftoff and translation operation are located relatively low on the spaceframe (see Figure 11-1), and therefore, are particularly susceptible to landing damage, especially from protruding rocks on the lunar surface. This problem is especially critical with respect to the vernier thrust chambers and the RADVS antennas. Finally, the crushable structure will, presumably, already have been crushed on landing the first time, and only reduced energy absorption capability will be available.

Thermal problems: It seems reasonalbe to conclude that the site certification operations must be conducted during the lunar day, so that sufficient lighting will be available for television surveys. Thus, if the original landing site is to be surveyed, followed by liftoff and translation to one or more additional sites where TV surveys are also to be taken, all of the required subsystems will need to operate in the daytime lunar thermal environment. The present Surveyor flight hardware (flight control, propulsion, and radars) is not qualified to operate reliably under such conditions; serious question exists that the vernier propulsion system or the RADVS will be capable of operating at all.

Therefore it may be concluded that using present subsystems and basic design, liftoff and translation of the Surveyor spacecraft along the surface of the moon does not appear to be a feasible means to accomplish the site certification mission.

MECHANIZATION CONSIDERATIONS AND GROUND RULES

The ground rules adopted for the liftoff and translation maneuver mechanization were based on a desire for minimum possible change to the Surveyor spacecraft. Hence, reviewing the required functions and the available subsystems on board Surveyor, the following conclusions emerge and were used as ground rules:

1) All boosting is to be performed by the existing vernier propulsion system, which is also to provide attitude control during the entire thrusting phase.

2) The existing flight control hardware is to be utilized wherever possible. Hence, commanded attitude maneuvers should be performed with the spacecraft under inertial control by torquing the rate gyros.

3) The descent to the lunar surface might reasonably be placed under control of the RADVS system; this would constitute a straightforward utilization of the identical functions used in the present descent system. Alternatively, an all-inertial mode is conceivable, incorporating integrating circuitry into the flight control electronics to dead reckon position and velocity. This approach was considered, but later discarded, due to the large errors inherent in such an open loop procedure. For the general range of parameter values which are found to be reasonable, the integrated thrust acceleration for the liftoff and translation maneuver would typically be in excess of 300 fps, nearly all of it in the vertical direction. Hence, a 1-degree misalignment of the vertical reference would lead to a horizontal velocity component due to this error alone, of greater than 5 fps, which is the presently specified allowable maximum. Since, as is discussed elsewhere in this section, the high temperature environment and the already-crushed state of the crushable structure imply a reduced horizontal velocity tolerance, descent utilizing all-inertial sensing can be ruled out on these grounds alone.

   It is apparent, though, that there are other significant problems associated with this guidance mode; in particular, since a closed-loop indication of altitude is not available, vernier engine shutoff at 13 feet altitude, as in the present radar guided system, is not practical. Accumulated altitude errors in a dead reckoning mode could easily lead to excessive touchdown velocities if engine shutoff above the moon's surface were attempted. One way to overcome this particular problem is to mechanize a constant velocity descent with engines burning until touchdown. Unfortunately, new problems are introduced by taking this approach; for example, the problem of dust, excited by the engine exhausts, impinging on the spacecraft. As a result of these reservations, it was decided not to consider further the all-inertial guidance mode.

   A third possibility for terminal descent sensing is the downward-looking TV camera already aboard the spacecraft. Investigation of this mode during the initial Surveyor proposal study phase indicated that the two-way propagation delay was sufficient to render such a descent control loop unstable. In addition, a self-tracking high gain antenna would be required for the wide-band TV transmission; thus, it was possible to quickly rule out this approach and to confine attention to utilization of the present descent sensing system.

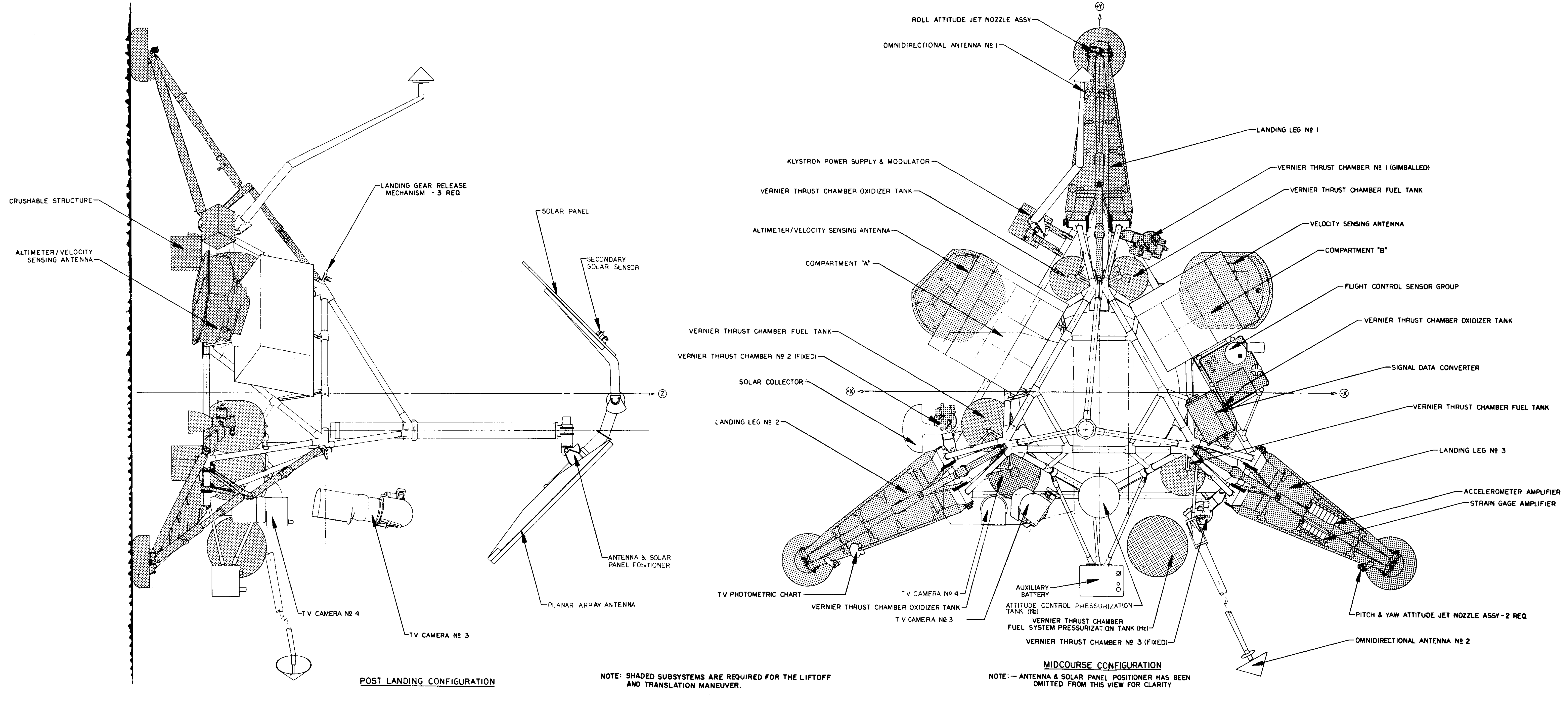


Figure 11-1. Surveyor Spacecraft General Arrangement

MANEUVER SEQUENCE

A liftoff and translation maneuver sequence was synthesized which is consistent with the ground rules and which is achievable at reasonable fuel costs. A summary of the sequence which was selected for study follows:

1) Preliftoff preparation. It is assumed that one or more television scans having been performed in the vicinity of the landed Surveyor spacecraft, it has been decided at the Space Flight Operations Facility (SFOF) that the spacecraft will be flown to a particular spot in the vicinity of the original landing site, as determined from the television frames. Spacecraft telemetry shows that all subsystems are operating as required to perform the maneuver. By means of prepared computer programs, the television data is processed to produce best estimates of the coordinates of the target landing spot in spacecraft coordinates. Using the telemetered outputs of a vertical gyro on board the spacecraft (new addition to the flight control system), which is brought into operation at this time, the coordinates of the target with respect to a local vertical coordinate system are also computed. Finally, the required commands for the liftoff and translation maneuver are computed. Consistent with the sequence described here, these are:

    a) Left or right roll

    b) Roll magnitude

    c) Pitch or yaw

    d) Plus or minus pitch/yaw

    e) Pitchover time magnitude

    f) Pitchback rate

    g) Radar acquisition time

   The required command tapes are punched and verified, and the commands transmitted to the spacecraft for storage in appropriate registers (new additions to the flight control system), then verified by telemetry return. Finally spacecraft subsystems required for the maneuver are turned on, and spacecraft readiness is verified from the telemetry.

2) Liftoff. The spacecraft is placed on inertial hold attitude control (pitch, yaw, and roll gyros in the integrating rate mode) and the vernier engines are ignited and accelerometer controlled to 1.2 g (lunar). A 1-second liftoff period is programmed, allowing the spacecraft to rise about 6 inches above the surface and to attain an upward velocity of about 1 fps.

3) Erection. Since the spacecraft may have been resting on a local slope as steep as 15 degrees, the initial liftoff thrusting may be misaligned from the local vertical by the same angle. Hence, after the 1-second liftoff phase, spacecraft pitch and yaw attitude control is switched by the flight control programmer to the vertical gyro reference; roll control is maintained by the roll gyro. Three seconds are allowed for the spacecraft roll axis to be brought to the local vertical, limiting pitch and yaw rates to the 5 deg/sec rate torquing capability of the gyros.

4) Roll to target azimuth. Since the translation maneuver requires that horizontal velocity be developed in the direction of the target, the spacecraft thrust axis will have to be shifted from the vertical to accomplish this. Since it is desirable to command an attitude maneuver about one principal axis only, it is necessary to roll the spacecraft to bring the axis, about which the off-vertical maneuver will be made, to a position normal to the spacecraft-to-target heading. To save fuel, the maximum roll maneuver which could be required is held to 45 degrees by allowing the off-vertical maneuver to be commandable as a pitch or yaw maneuver in either the positive or negative sense in each. Also as a fuel saving measure, spacecraft thrust acceleration during the roll maneuver is switched to 1 g (lunar) by the flight control programmer, thus limiting vertical velocity buildup. Alternatively, the 1 g thrusting may be accomplished more directly and more accurately, at the expense of a relatively minor RADVS circuit change, by controlling the spacecraft to a constant upward velocity during roll. The maneuver is accomplished by torquing the roll gyro from a precision calibrated current source corresponding to a 5 deg/sec rate for a time interval controlled by the magnitude commanded into the roll register. Hence, the maximum required roll maneuver is of 9 seconds duration.

5) Pitchover. After completion of the roll maneuver, the thrust acceleration is switched back to the 1.2-g (lunar) level, and the 5-deg/sec torquing current is switched to the appropriate pitch or yaw maneuver axis, while the spacecraft clock starts to count down the register containing the stored pitchover time magnitude command. When the register reaches zero, the pitchover is completed. As shown later, the pitchover time magnitude is variable and is based on the translation distance desired.*

6) Coast and pitchback. When the pitchover maneuver has been completed, the engines are throttled back to 0.9-g (lunar) thrust acceleration, a level which is a practical lower limit consistent with the Surveyor Block I vernier engine thrust range. Simultaneously, the pitch (yaw) rate is switched from the constant 5-deg/sec rate to a rate of the opposite sign, lower in magnitude, which was previously computed on the ground, commanded, and stored as the pitchback rate. As shown later, the magnitude of the pitchback rate is variable with the translation distance desired, and is such that when the spacecraft reaches the apex of its trajectory (i.e., zero vertical velocity), the spacecraft roll axis will again be vertical. This implies that a mirror image of the attitude and thrust profiles (with the exception of the 1 g thrust phase roll maneuver), would serve, on a completely regular lunar surface and with an error-free system, to bring the spacecraft to a landing after having translated twice the distance from the original landing site to the apex of the trajectory. This symmetrical property of the trajectory could be utilized in an inertial guidance mode if such a scheme were practical in this application, but as discussed earlier, the inertial mode has been discarded as not feasible.

7) Radar acquisition. After the apex of the trajectory has been passed, the continuing vehicle pitch rate begins to carry the vehicle thrust axis away from the local vertical again, starting the removal of horizontal velocity. At the same time the flight path angle, which was horizontal at the apex, begins to turn toward the vertical as the spacecraft gains downward velocity. The thrust axis and the velocity vector approach each other, and at some point along the trajectory, precomputed on the ground as the radar acquisition time, they will coincide. At this commanded time, the spacecraft is switched to radar control, and attitude and thrust are thereafter controlled as in the original descent. From this point on the two halves of the

*"Pitchover" is used here generically to denote the off-vertical attitude maneuver which may be about either the pitch or the yaw axis.

trajectory are no longer mirror images of each other, and the distance from the liftoff site to the landing site will not be twice the distance from the liftoff site to the apex. This biasing of the trajectory must be incorporated into the command computation on the ground.

In general, accumulated errors will cause a finite angular error between the spacecraft roll axis and the velocity vector at the instant of switchover to radar control. This error will be handled in precisely the same way that the analogous (and larger) burnout error is handled in the original descent: the spacecraft will be slewed, under doppler radar control, to remove the angular error.

LIFTOFF AND TRANSLATION ANALYSIS

A simple model of the trajectory problem is shown in Figure 11-2. Action in the trajectory plane only is considered. For the sake of simplicity, no slope at the liftoff point is assumed; therefore there is no horizontal component of thrust during the liftoff period (t_0 to t_1). The pitchover takes place from t_1 to t_2, the pitchback from t_2 to t_3 and beyond, and radar acquisition is at t_4. The horizontal displacement at t_3 is shown as d/2. Thus, d is the distance which would be traversed were radar steering not applied at t_4, and is not the true translation distance. For rough assessment purposes, however, d is an adequate approximation to the true distance and it will be considered as such.

The trajectory analysis is extremely straightforward, and is summarized below. Impulsive angular velocities are assumed throughout because they are an extremely good approximation when vernier engine actuated attitude control is employed.

Phase I (t_0 to t_1)

Assume that thrust acceleration is a_{max}. Assume $t_0 = 0$.

$$\ddot{x}(t) = 0 \qquad \dot{x}(t) = 0 \qquad x(t) = 0$$

$$\dot{\theta}(t) = 0 \qquad \theta(t) = 0$$

$$\left.\begin{aligned} \ddot{y}(t) &= a_{max} - g \\ \dot{y}(t) &= (a_{max} - g)\,t \\ y(t) &= \frac{1}{2}(a_{max} - g)\,t^2 \end{aligned}\right\} \quad 0 \le t \le t_1$$

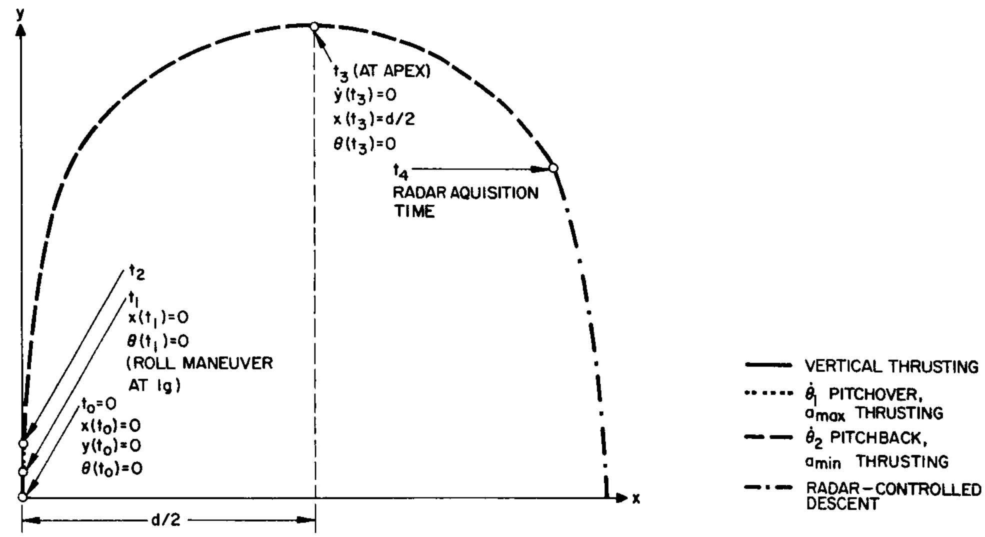


Figure 11-2. Liftoff and Translation Maneuver

Phase II $(t_1$ to $t_2)$

Thrust acceleration = a_{max}

Pitch rate = $\dot{\theta}_1$

$$\theta(t) = \dot{\theta}_1 (t - t_1)$$

$$\therefore \theta(t_2) = \dot{\theta}_1 (t_2 - t_1)$$

$$\ddot{x}(t) = a_{max} \sin \theta(t)$$

$$\dot{x}(t) = \int_{t_1}^{t} a_{max} \sin \theta_1 (\tau - t_1)\, d\tau = \frac{a_{max}}{\dot{\theta}_1} \left[1 - \cos \dot{\theta}_1 (t - t_1)\right], \quad t_1 \leq t \leq t_2$$

$$\therefore \dot{x}(t_2) = \frac{a_{max}}{\dot{\theta}_1} \left[1 - \cos \dot{\theta}_1 (t_2 - t_1)\right]$$

It also follows that

$$x(t) = \frac{a_{max}}{\dot{\theta}_1} \left[(t - t_1) - \frac{\sin \dot{\theta}_1 (t - t_1)}{\dot{\theta}_1}\right], \quad t_1 \leq t \leq t_2$$

$$x(t_2) = \frac{a_{max}}{\dot{\theta}_1} \left[(t_2 - t_1) - \frac{\sin \dot{\theta}_1 (t_2 - t_1)}{\dot{\theta}_1}\right]$$

In the vertical direction

$$\ddot{y}(t) = a_{max} \cos \dot{\theta}_1 (t - t_1) - g$$

with velocity and position relationships following directly. The above expressions were derived but were unwieldy for the quick hand-computed results which were desired, and became especially unwieldy in the next phase during pitchback. As a result, for the generally small angles involved here, the approximation in the vertical direction of $\cos \theta = 1$ was employed. Hence, the vertical analysis, in all phases, is quite simple:

$$\dot{y}(t_1) = (a_{max} - g)\, t_1$$

$$y(t_1) = \frac{1}{2}(a_{max} - g)\, t_1^2$$

$$\dot{y}(t_2) = (a_{max} - g)\, t_2$$

$$y(t_2) = \frac{1}{2}(a_{max} - g)\, t_2^2$$

<u>Phase III (t_2 to t_3)</u>

Thrust acceleration = a_{min}.

Pitch rate = $-\dot{\theta}_2$.

For $t_2 \leq t \leq t_3$:

$$\dot{y}(t) = (a_{max} - g)\, t_2 + (a_{min} - g)\,(t - t_2)$$

$$= (a_{max} - a_{min})\, t_2 + (a_{min} - g)\, t$$

thus

$$\dot{y}(t_3) = 0 = (a_{max} - a_{min})\, t_2 + (a_{min} - g)\, t_3$$

$$t_3 = t_2 \frac{a_{max} - a_{min}}{g - a_{min}} \tag{11-1}$$

$$y(t) = y(t_2) + \int_{t_2}^{t} \dot{y}(\tau)\, d\tau$$

$$= (a_{max} - a_{min})(t - t_2/2)\, t_2 + t^2/2\, (a_{min} - g)$$

Also in Phase III:

$$\theta = \dot{\theta}_1 (t_2 - t_1) - \dot{\theta}_2 (t - t_2)$$

$$\therefore \quad \theta(t_3) = 0 = \dot{\theta}_1 (t_2 - t_1) - \dot{\theta}_2 (t_3 - t_2)$$

and

$$\dot{\theta}_2 = \dot{\theta}_1 \frac{t_2 - t_1}{t_3 - t_2} \tag{11-2}$$

Also

$$\dot{x}(t) = \dot{x}(t_2) + \int_{t_2}^{t} a_{min} \sin\theta(\tau)\, d\tau$$

$$= \frac{a_{max}}{\dot{\theta}_1} \left[1 - \cos\dot{\theta}_1 (t_2 - t_1) \right] + \frac{a_{min}}{\dot{\theta}_2} \left\{ \cos\left[\dot{\theta}_1 (t_2 - t_1) - \dot{\theta}_2 (t - t_2) \right] \right.$$

$$\left. - \cos\dot{\theta}_1 (t_2 - t_1) \right\}$$

$$\therefore \quad \dot{x}(t_3) = \left[1 - \cos\dot{\theta}_1(t_2 - t_1)\right]\left[\frac{a_{max}}{\dot{\theta}_1} + \frac{a_{min}}{\dot{\theta}_2}\right]$$

$$x(t) = \frac{a_{max}}{\dot{\theta}_1}\left[(t_2 - t_1) - \frac{\sin\dot{\theta}_1(t_2 - t_1)}{\dot{\theta}_1}\right]$$

$$+ (t - t_2)\left\{\frac{a_{max}}{\dot{\theta}_1}\left[1 - \cos\dot{\theta}_1(t_2 - t_1)\right] - \frac{a_{min}}{\dot{\theta}_2}\cos\dot{\theta}_1(t_2 - t_1)\right\}$$

$$- \frac{a_{min}}{\dot{\theta}_2^2}\left\{\sin\left[\dot{\theta}_1(t_2 - t_1) - \dot{\theta}_2(t - t_2)\right] - \sin\dot{\theta}_1(t_2 - t_1)\right\}$$

Finally, the following may also be obtained:

$$x(t_3) = \frac{d}{2} = \frac{a_{max}}{\dot{\theta}_1}(t_3 - t_1) + \left[\sin\dot{\theta}_1(t_2 - t_1)\right]\left[\frac{a_{min}}{\dot{\theta}_2^2} - \frac{a_{max}}{\dot{\theta}_1^2}\right]$$

$$- (t_3 - t_2)\left[\cos\dot{\theta}_1(t_2 - t_1)\right]\left[\frac{a_{max}}{\dot{\theta}_1} + \frac{a_{min}}{\dot{\theta}_2}\right] \tag{11-3}$$

The foregoing relationships are sufficient to calculate the trajectory parameters of interest. A reasonable approximation to the actual propellant requirements may be obtained from the simple symmetrical model approximation. Let ΔV_{eq} be the total equivalent boost velocity requirement, i.e., the total integrated thrust acceleration. Then

$$\frac{\Delta V_{eq}}{2} = g t_{roll} + a_{max} t_2 + a_{min}(t_3 - t_2)$$

where t_{roll} = time interval required to roll to the target heading. Maximum t_{roll} = 9 seconds.

Finally, for comparison purposes the ideal velocity requirements for liftoff and translation may be easily derived. Consider impulsive velocity increments on a flat moon. Then, referring to Figure 11-3, the ideal velocity requirement would result from an impulsive boost at a 45-degree flight path angle, a thrust-free coast period, followed by an impulsive deboost at 45 degrees. For this situation, the relationship for the total required boost velocity is:

$$\Delta V = 2 \sqrt{gd}$$

This expression is useful for comparison purposes, as shown in the following section.

TRAJECTORY AND PERFORMANCE RESULTS

As may be seen from the results presented in the above discussion, the following parameters must be fixed to define a particular trajectory:

$$t_1, t_2, t_3, a_{max}, a_{min}, \dot{\theta}_1, \dot{\theta}_2, d$$

To determine these eight unknowns, only three constraining relationships have been obtained: Equations 11-1 through 11-3, which correspond to the specifications at the apex that the vertical velocity is zero, the roll axis is vertical, and half the horizontal distance has been traversed.

Thus, 5 degrees of freedom remain. One is necessary to allow a parametric variation of distance traversed. Hence, four values of parameters must either be selected on the basis of equipment limitations or optimized in some sense.

a_{min} should clearly be as low as possible, since ideally no thrust acceleration at all is desired. From the standpoint of practical mechanization, 0.9 g (lunar) is about all that is achievable under Surveyor Block I constraints; for Block II, improved engines could conceivably allow some lowering. For calculation purposes, 0.9 g was selected.

$\dot{\theta}_1$ should be relatively large, to provide a rapid buildup of horizontal velocity. Since 5 deg/sec is the present torquing limit of the Surveyor gyros, this value was selected.

For a_{max}, the present average maximum acceleration in descent of 2.08 g was selected initially. However, the fuel costs at this acceleration level are excessive, so the more practical value of 1.2 g was selected for later calculations.

Finally, T_1 was selected as 4 seconds as described earlier.

Several figures are presented to show the results. Figure 11-4 shows boost velocity (and fuel) requirements as functions of the translation distance. In earlier computations, 14 seconds of thrusting at 2.08 g were assumed, allowing for initial liftoff, erection, and a maximum roll angle of 45 degrees. For this reason, the thrust cutback to 1 g during roll was decided on. The highest curve shows the results where t_1 = 4 seconds, and the roll maneuver is performed with throttle-back to 1 g, while a_{max} = 2.08; the lowest curve shows the requirements in the ideal case. Clearly, the penalties of this mechanization scheme are very high, and are also almost independent of translation distance. Thus, it is the velocity loss implicit in the original liftoff scheme which must be reduced.

a_{max} was reduced to 1.2 g, deemed as low a value as is feasible if the initial liftoff from the surface is to be achieved cleanly and safely. The middle curve shows the results for this method, where the velocity penalty for mechanization is more than cut in half.

Figure 11-5 shows a typical trajectory for the upper curve of Figure 11-4, where a_{max} = 2.08 g. Note, the steepness of the trajectory. Figure 11-6 shows a family of trajectories for the maneuver sequence finally selected (t_1 = 4 seconds, a_{max} = 1.2 g, a_{min} = 0.9 g, $\dot{\theta}_1$ = 5 deg/sec), with varying values of t_2, and hence, of all the other parameters including translation distance. The velocity costs of these trajectories may be found from the middle curve of Figure 11-4.

The results of this section show that translation over significant distances may be achievable at reasonable fuel costs. For example, approximately 620 feet may be traversed at a fuel cost of 4 percent of liftoff weight (based on an I_{sp} of 285 seconds) equivalent to 375 fps of boost velocity. Thus a 700-pound spacecraft could make the maneuver while using less than 30 pounds of fuel, and might even make two such successive maneuvers for less than 60 pounds of vernier fuel.

FEASIBILITY CONSIDERATIONS

It was pointed out earlier that the practical problems of environment and reliability probably preclude the feasibility of implementing a liftoff and translation capability for Surveyor Block II. These considerations fall into two general areas: 1) structural and dynamics, and 2) thermal.

Structural and Dynamic Problems

Referring back to Figure 11-1, the critical components and subsystems needed for the liftoff and translation maneuver are shaded. Note that these, unfortunately, are mostly concentrated toward the lower part of the spacecraft, where they are most susceptible to landing damage. The systems

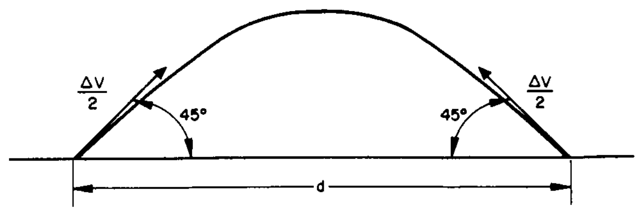


Figure 11-3. Ideal Liftoff and Translation

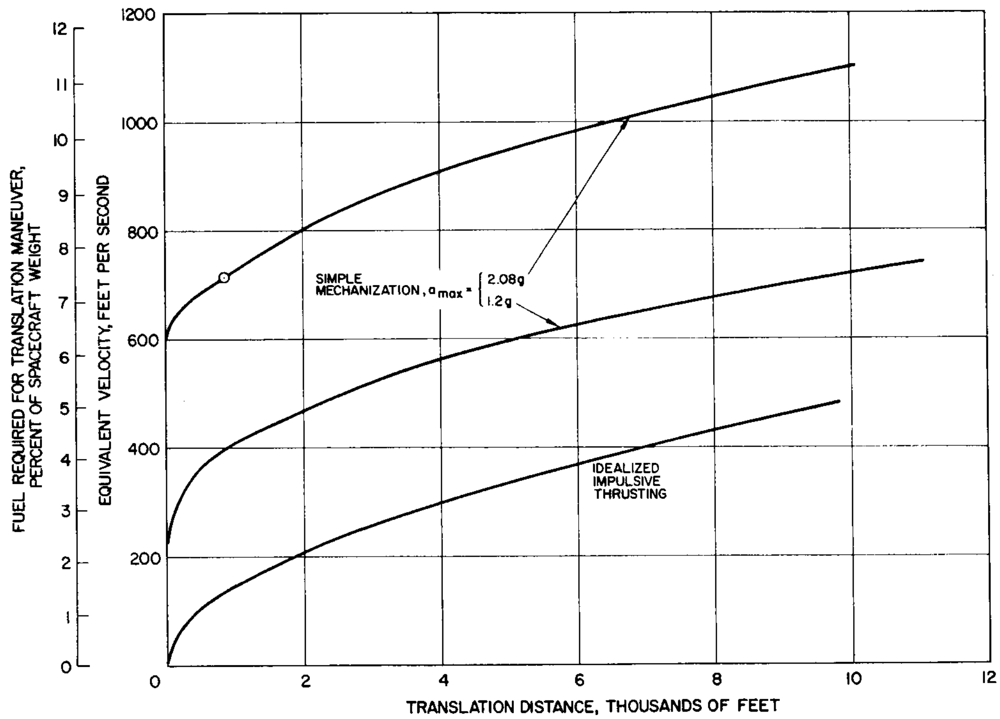


Figure 11-4. Velocity and Fuel Cost versus Translation Distance

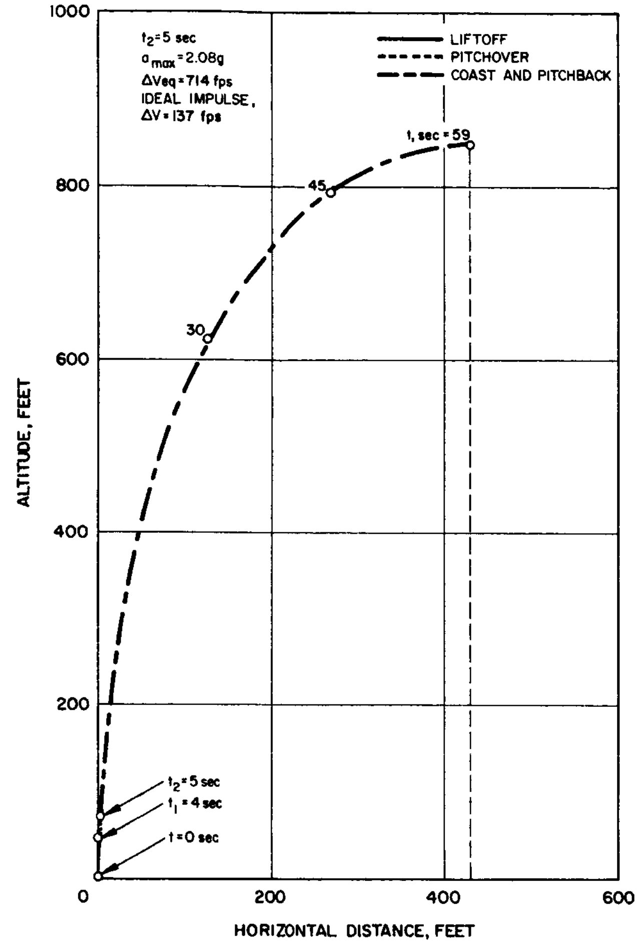


Figure 11-5. Typical Trajectory Profile

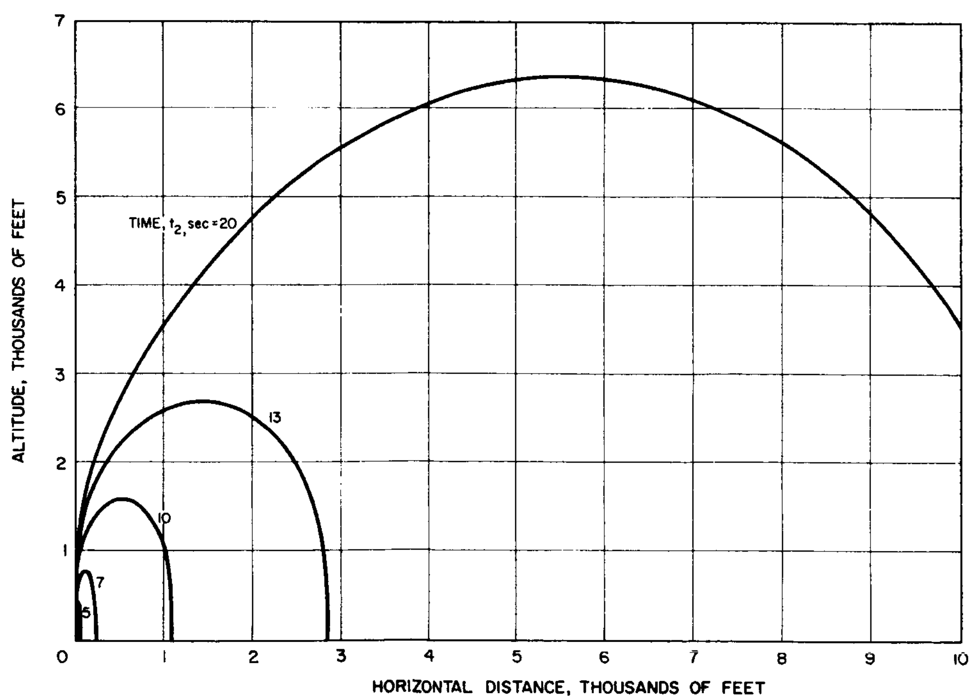


Figure 11-6. Trajectory Family for Selected Maneuver Sequence

involved are the vernier engines, the RADVS system, the flight control system, and the landing gear, shock absorbers, foot pads, and crushable structure. It is no accident that these subsystems are so located, since some of them must be located there by function (verniers, RADVS, crushables, gear), and further, Block I Surveyor does not require them to function after landing. Hence, it was not necessary to locate them so as to minimize postlanding damage. Of particular interest is the fact that the crushable structure may be crushed essentially to the exit plane of the vernier engine nozzles and to the lowest points of the RADVS antennas, placing these components in an extremely vulnerable position with respect to surface protuberances. Therefore, from the point of view of protection to the crucial subsystems required, the Surveyor spacecraft is not well designed for the liftoff and translation maneuver.

Of additional concern is the fact that at second touchdown, the already-crushed crushable structure will be unable to perform the vital energy absorbing function. The result will be overloading and probable bottoming of the shock absorbers, transmitting overloads into the spaceframe. Thus structural failure due to peaked loads is a distinct possibility. In addition, the already-crushed crushables will be unable to contribute to landing stability from the standpoint of toppling resistance. Thus, the overall result is a much reduced confidence in the landing system capability.

Thermal Problems

Liftoff and translation feasibility is also called into question by the thermal environment within which the subsystems may be called on to operate. Since it is not possible to assume that the maneuver may not be called for at any time during the lunar month, the capability of the pertinent spacecraft subsystems was estimated for the worst case lunar environment, occuring for most of them within one or two days of lunar noon. The results are indicated here:

1) Flight control system. The thermal environment is probably least harmful in the case of flight control. The flight control electronics is estimated to be in the region of 260 to 270°F, where the major effect will be lower reliability. The roll actuator is expected to be at approximately 250°F, also causing reduced reliability. The Inertial Reference Unit (IRU) can be expected to be at about 150 to 200°F, but since the normal gyro operating temperature is 180°F, no particular performance degradation is expected.

2) RADVS. The effect here is extremely severe. The antennas may become as hot as 500°F, at which point some of the plastic materials will outgas and very probably fail. The klystron power supply will be at 300 to 350°F, and the addition of operating power will very likely produce thermal failure.

3) Vernier propulsion system. The estimated vernier component temperatures are as follows:

Thrust chambers and valves:	250° F
Tanks (fuel, oxidizer, helium):	200 to 250° F
Fuel and oxidizer lines:	250 to 300° F

At these temperatures, it is doubtful that the vernier system can perform. The coil resistance in the propellant valves and torque motors will be raised sufficiently high that functioning is questionable. The inlet temperatures may cause injector failure. With the present propellants, the oxidizer will be completely gaseous at all times, while the fuel will be gaseous at all pressures below 50 psi, corresponding to the midthrust point. Under these conditions, combustion is attainable, but startup time and transient response become very poor. Some test results show response times as long as 1 second to several seconds. Under the gaseous flow conditions, the injector will become the limiting orifice, and the throttle valve will be saturated. In the case of engines with a cavitating venturi, the mixture ratio will be severely disturbed. The problem is one of the basic properties of the fuel and oxidizer used; it cannot be solved by a minor chamber design change. In conclusion, the vernier engines cannot function under the conditions cited.

4) Shock absorbers. The shock absorber temperatures may go as high as the 250 to 325° F range. The shock absorbers could not then be expected to perform properly. Leaks may occur; in addition the spring and damping constants would be badly mismatched, with a deleterious effect on landing stability.

Based on the foregoing discussion it is clear that the current spacecraft subsystem designs are not compatible with the maneuver requirements.

CONCLUSIONS

From the relatively cursory study of the liftoff and translation maneuver which has been performed, it is possible to draw several basic conclusions. From a conceptual and basic system performance standpoint, the liftoff and translation maneuver is a perfectly reasonable mission for a Surveyor-type spacecraft. Almost all of the elements required to perform the mission are already aboard the spacecraft to satisfy the original landing requirements. Only small additions of instrumentation (vertical gyro) and control circuitry are necessary to provide all of the elements needed to implement the liftoff and translation maneuver.

Viewed from the standpoint of tradeoff between performance and mechanization complexity, a very simple scheme is possible, implementing the maneuver at moderate cost in fuel/velocity penalty. This penalty is almost constant at approximately 250 fps regardless of translation distance. Thus for comparison, 1000 feet could be traversed by an ideal system at a cost of approximately 150 fps, while the method used here would involve a boost velocity expenditure of about 400 fps. Though the ratio of velocities is high, even the higher figure may be realized at the cost of less than 4-1/2 percent of spacecraft weight, or about 30 pounds of vernier propellant for a 700-pound landed spacecraft weight.

Despite the moderately favorable conclusions above, even a comparatively brief review of the environmental and reliability aspects leads inescapably to the conclusion that little confidence could be placed in the capability of a Surveyor spacecraft to perform the liftoff and translation maneuver. The landing damage and high temperature lunar surface problems would make success unlikely.

12. PROGRAM PLAN

A program plan for Block II Surveyor has been developed in consonance with the restraints imposed by the 1 July 1964 go-ahead (Figures 12-1 and 12-2). An initial February 1967 launch, with subsequent launches every 2 months, is planned. Based on a 1 July 1964 go-ahead, including the scientific payload definition by JPL, the spacecraft configuration will be defined by 1 September. Development of the modified control items will be completed by 1 March 1965 at which time the design freeze with drawing release will occur.

An additional structural test vehicle, S-2A, will be assembled. Completion of the S-2A test program is scheduled for 1 July 1965 The T-2 dynamic descent vehicle will be upgraded to the 2600-pound* configuration incorporating the new vernier system. Testing of this system will be completed by 1 July 1965.

Final assembly of the T-26 prototype vehicle will be completed by mid-October 1965 and will undergo a 3-1/2 month period of system functional tests. Type approval environment consisting of vibration and drop testing of T-26 will occur through mid-March 1966, followed by 2 months of thermal-vacuum tests.

Units for control item TAT will be delivered 1 December 1965, and the SC-II-1 (first of the Block II flight spacecraft) units will be delivered by 1 May 1966. Assembly of SC-II-1 will be completed 1 July. SC-II-1 will then undergo 2-1/2 months of system functional testing followed by 1 month each of vibration and thermal-vacuum testing. The flight spacecraft will be shipped to GDA for combined system tests for 1 month and then delivered to AMR for 2 months of checkout prior to launch.

Development of a retro rocket engine of the chosen configuration for flight spacecraft will occupy a 13-1/2 month span ending mid-October 1965. Qualification of the new engine will be completed 15 March 1966 with delivery of a flight model engine by mid-December 1966.

*A 2600-pound injected weight is used for reference only; the program plan is not sensitive to injected weight provided that units requiring change are as described herein.

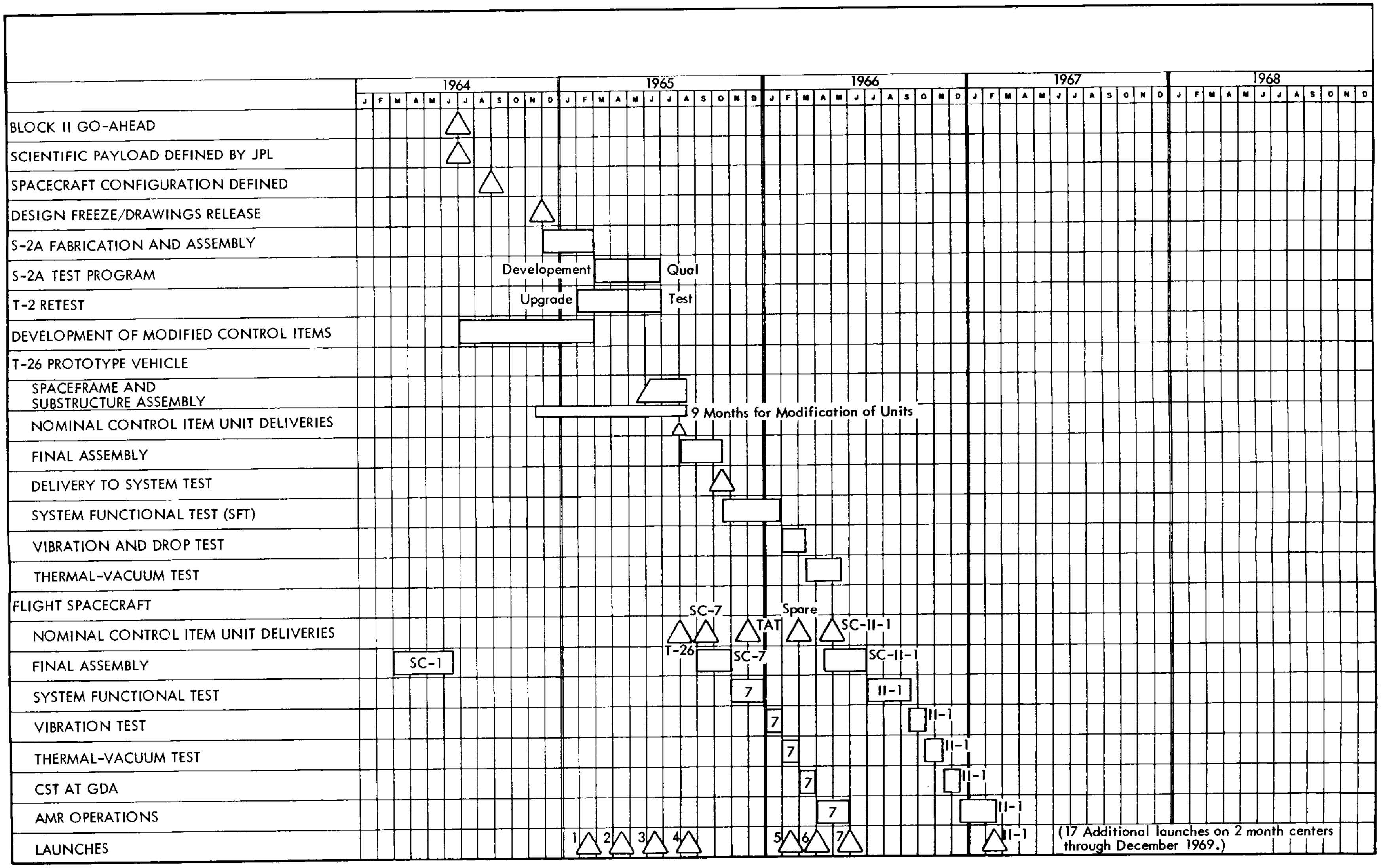


Figure 12-1. Program Plan for Block II Surveyor

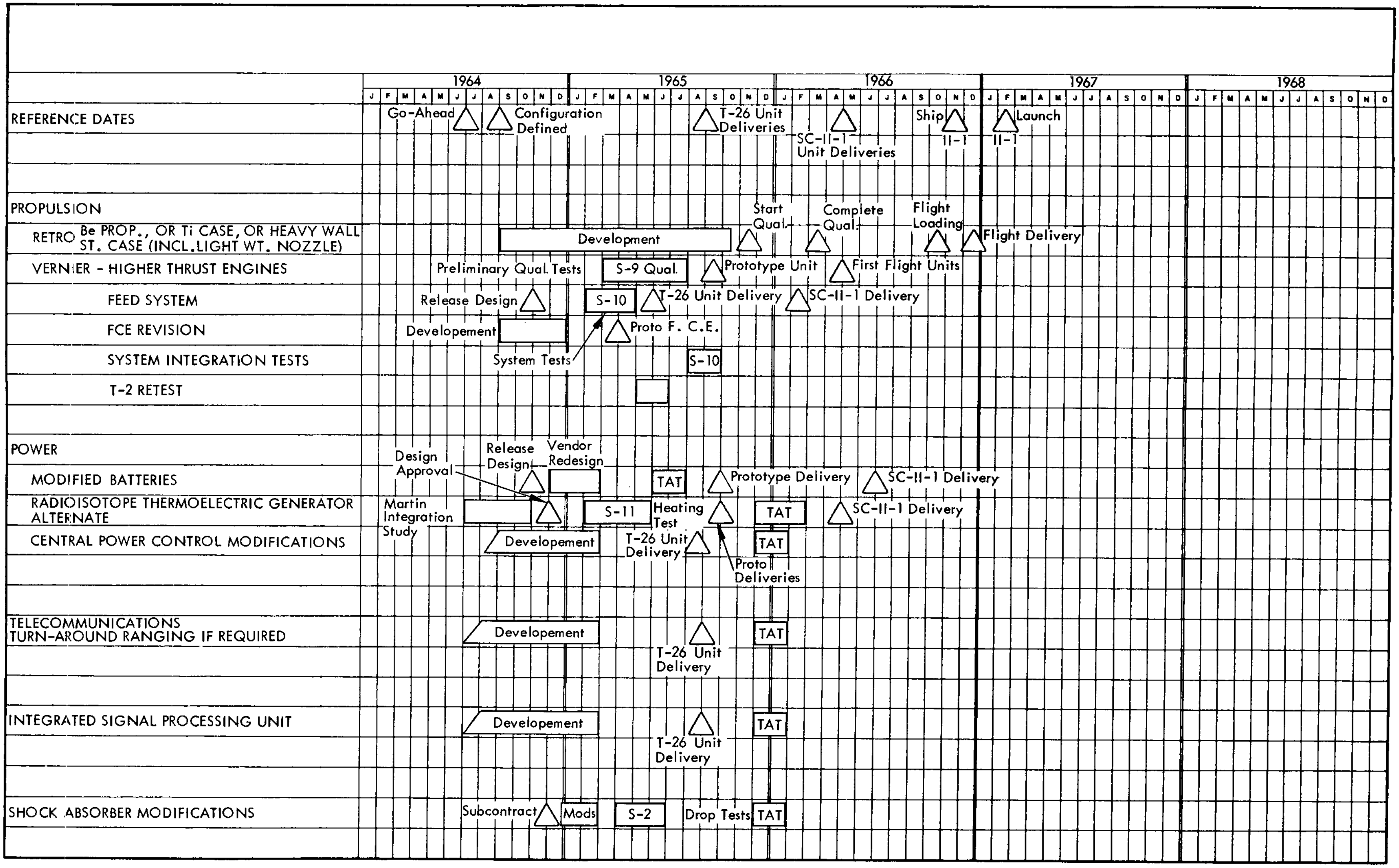


Figure 12-2. Program Plan for Block II Surveyor Subsystem Development

Qualification of the higher thrust vernier engines on the S-9 vehicle will be completed on 1 August 1965 with delivery of the prototype units on 15 September. The first flight model engines are scheduled for delivery on 1 May 1966.

The feed system, modified as a result of the new vernier engines, will undergo a 3-month system test ending 1 May 1965 while the prototype unit will be delivered 1 month later. The flight model feed system will be delivered mid-February 1966.

Revisions to the flight control electronics unit to incorporate the requirements of the vernier engine system will be completed 1 January 1965, and a 2-month system integration test utilizing S-10 will be performed through September 1965 utilizing a prototype electronics unit.

Battery redundancy will be added to the power system. Design release of the modified battery is scheduled for 1 November 1964 with the redesign being completed by 1 March 1965. The batteries will be subjected to TAT during the months of June and July with prototype units being delivered 1 October 1965. The radioisotope thermoelectric generator, a power alternate, design approval is scheduled for 1 December 1964 with heating tests conducted on the S-11 vehicle. Prototype delivery is anticipated 1 October 1965 with TAT scheduled for completion 1 March 1966.

Modification of the central power control unit, telecommunications unit (if ranging is incorporated), and the integrated signal processing unit will be completed 1 March 1965. Delivery of the T-26 units are scheduled for 1 September 1965. Type approval testing of these units will be completed by 1 February 1966.

The shock absorbers for the Surveyor spacecraft will be modified for the 2600-pound configuration. The new units will be placed in a drop test program in April 1965 with completion scheduled in June. Type approval testing will be performed in December and January 1966. Delivery of flight hardware will be made in August of 1966.

All prototype and TAT units will be produced by the Hughes engineering divisions (with the exception of subcontract items). Spare and SC-II-1, plus all subsequent units, will be manufactured at the Hughes El Segundo production facility. The spare unit will be utilized to proof the production operations at the El Segundo facility prior to being used as a spare to the SC-II-1 spacecraft. The quantity of spacecraft considered for Block II justifies the use of production line techniques. Units that require no design changes will be phased into the production facility at the earliest possible date.

The effects of the recent change of vernier engine subcontractor have not been included in the program plan.

13. PLANNING PURPOSE FUNDING REQUIREMENT

Fiscal year funding requirements for a Block II Surveyor program as described in Section 12 of this report are estimated as follows:

Fiscal Year	Planning Purpose Funding Requirement	
1965	$ 33,000,000	
1966	32,000,000	
1967	35,000,000	
1968	30,000,000	
1969	25,000,000	
1970	13,000,000	
Total	$168,000,000	(at G &A level)

The developmental phase of the Block II program is considered to end with the completion of TAT of the T-26 prototype vehicle, 22-1/2 months after the start of the program. Extrapolation of A-21/A-21A experience leads to an approximate cost for this phase of $50 million. Included in this figure is increased subcontract cost of $6. 5 million, primarily for development and qualification of propulsion for the heavier vehicle; this figure is subject to wide variation as a function of the propulsion configuration that is chosen. The remaining $43. 5 million cost of the development phase represents Hughes in-house manpower and nonlabor costs. The Block II spacecraft configuration is not determined at this time, but the tasks to be performed are sufficiently well understood that this estimate is considered accurate for planning purposes.

A typical cost figure for an A-21A spacecraft, including production, test and operations, is $5 million. For the early Block II flights, this figure will be higher, but in view of the total number of expected flights, some benefit from a learning curve can be realized during the course of the production program. Consequently, there appears to be no valid reason for changing from the estimated cost of $5 million per flight spacecraft. Thus, the estimated cost for production and operations is $90 million.

The quantity of spacecraft considered for Block II justifies the use of factory production techniques. The cost of factory implementation, by comparison with other programs, is estimated at $8 million. This should not be considered an additional program cost, for it is believed that savings equivalent to the factory implementation cost will be achieved during the course of the production run; the cost per spacecraft would be higher if all spacecraft were produced by the engineering divisions. Implementation costs will start being incurred during February 1965.

After completion of development, sustaining engineering costs will be incurred in support of production and test. By extrapolation of A-21 experience to date, this is estimated at $20 million.

The following is a summary of planning purpose costs by program phase:

Program Phase	Planning Purpose Cost
Development	$ 50,000,000
Production implementation	8,000,000
Production and operations	90,000,000
Support engineering	20,000,000
Total	$168,000,000

These planning purpose costs assume that all 18 Block II Surveyors will be the same, and do not include costs for development of the payload itself.

MANPOWER

The anticipated manpower level for the Block II program is shown in Figure 13-1. The same figure shows the comparison of the manpower level estimated for the remainder of the A-21/A-21A program. The curve depicting total manpower for the two programs shows a smooth transition between programs.

The effects of the recent change of vernier engine subcontractor are not included in these estimates.

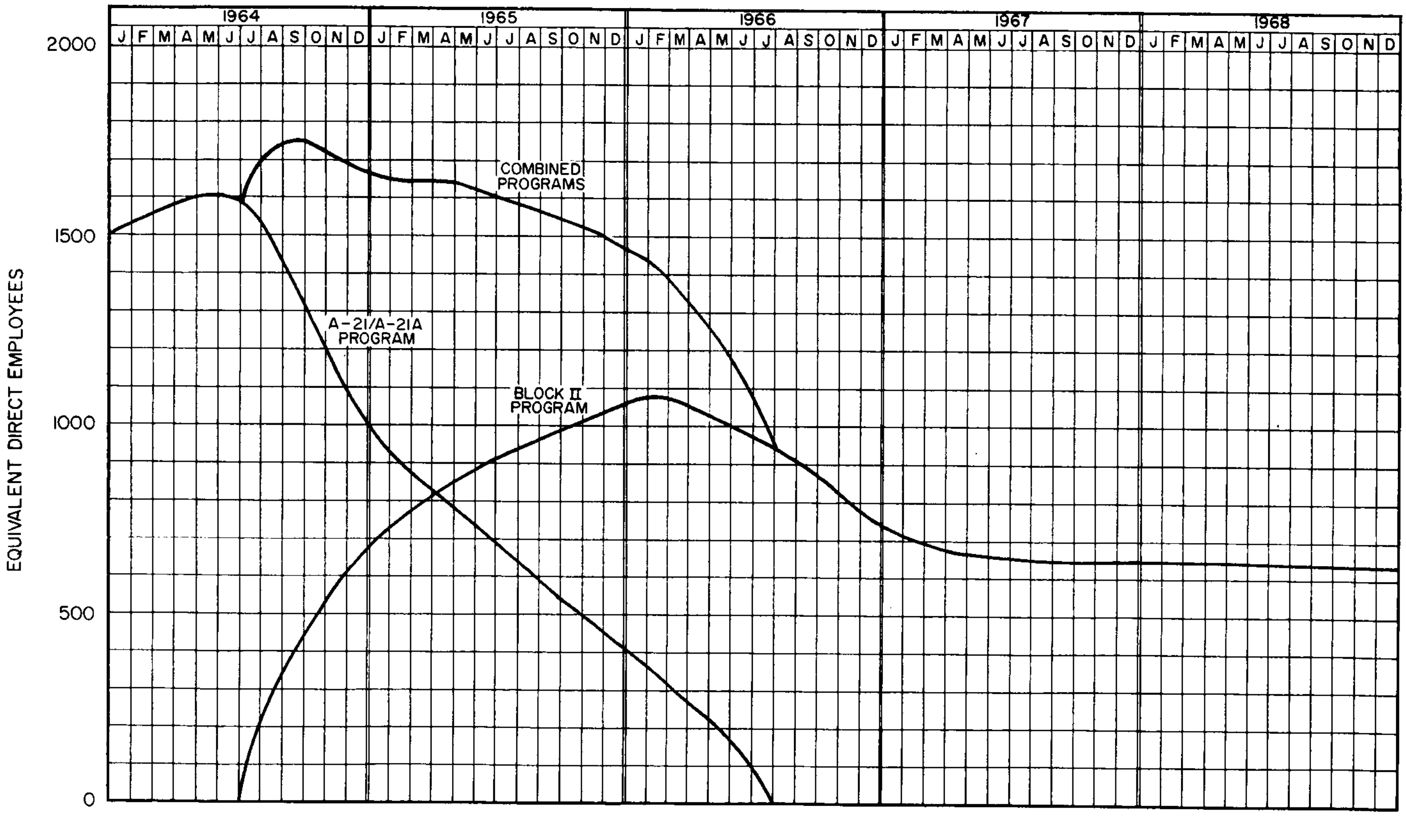


Figure 13-1. Manpower Requirements

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